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AIRCRAFT COMPANY, INC.

REPORT NUMBER

E.S. 17673

TECHNICAL REPORT ON HIGH ALTITUDE
AND HIGH SPEED STUDY

COPY # 28

EL SEGUNDO, CALIFORNIA, U. S. A.



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ES 17673

DOUGLAS AIRCRAFT COMPANY, INC. EL SEGUNDO DIVISION EL SEGUNDO, CALIFORNIA

1.0 SUMMARY

Results of the study are shown in "Summary Report High Altitude and High Speed Study", Report No. ES 17657.

DOUGLAS AIRCRAFT COMPANY, INC. EL SEGUNDO DIVISION EL SEGUNDO, CALIFORNIA

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3.0 INTRODUCTION

The purpose of the high altitude study which is summarized in this report is to establish the feasibility of extending human flight boundaries to extreme altitudes, and to investigate the problems connected with the design of an airplane for such flights.

This project is partially a result of man's eternal desire to go higher, faster, or farther than he did last year. Of far more importance, however, is the experience gained in the design of aircraft for high-speed, high-altitude flight, the collection of basic information on the upper atmosphere, and the evaluation of human tolerance and adaptation to the conditions of flight at extreme altitudes and speeds.

The design of an airplane for such a purpose cannot be based on standard procedures, nor necessarily even on extrapolation of present research airplane designs. Most of the major problems are entirely new, such as carrying a pilot into regions of the atmosphere where the physiological dangers are completely unknown, and providing him with a safe return to earth. The type of flight resembles those of hypersonic, long-range, guided missiles currently under study, with all of their complications plus the additional problems of carrying a man and landing in a proper manner.

Very little other work is available which deals wholly with the subject of the present study. Some theoretical studies on satellite vehicles are applicable in part; some of the work on long-range missiles is useful, although the basic concepts are often considerably different; and much of the aero-medical research being conducted for application to future fighter designs can be adapted to provide insight on certain phases of the present design.

The study which is summarized in the following sections consists of a first approach to the design of a high-altitude airplane. It attempts to outline most of the major problems and to indicate some tentative solutions. As with any preliminary investigation into an unknown regime, it is doubtful that adequate solutions have been presented to every problem of high-altitude flight, or even that all of the problems have been considered. It would certainly appear, however, that the major difficulties are not insurmountable.

4.0 CONFIGURATION

A three-view drawing of a proposed configuration for the high altitude study is shown in Figure 1. It is by no means implied that this represents a detailed design, but the configuration shown should be considered as a first approach to the problem. It is probable that detailed investigations of the performance, stability, aerodynamic heating, and general flight characteristics will produce appreciable changes in detailed design components, and perhaps even in the basic arrangement.

A conventional configuration was deliberately chosen for study, and no benefits have yet been discovered for any unconventional arrangement. Actually, for the prime objective of attaining very high altitudes, the general shape of the airplane is relatively unimportant. Stability and control must be provided, and it must be possible to create sufficient lift for the pullout and for landing; but, in contrast to the usual airplane design, the reduction of drag is not a critical problem and high drag is to some extent beneficial.

The planform of the wing is unimportant from an aerodynamic standpoint at the higher supersonic Mach numbers. Therefore it was possible to select the planform on the basis of weight and structure and landing conditions. These considerations led to the choice of an essentially unswept wing of moderate taper and aspect ratio. A more detailed study of other applications for the airplane, which might involve sustained flight at low supersonic speeds, would probably indicate the desirability of increasing the sweepback of the wing.

The fuselage is arranged around the propellant tanks, which are proposed as integral parts of the structure.

The tail surfaces are of proper size for stability at the lower supersonic Mach numbers, but there is some question of their adequacy at very high supersonic speeds. Further experimental data in this speed range are necessary before modifications are attempted. In addition, it may be possible to accept a certain amount of instability with the proper automatic servo controls.

Generally, for performing a given mission a two-stage vehicle is more efficient than a single-stage. The present configuration might be considered a two-stage aircraft because the manned aircraft is carried to 40,000 feet by a mother ship before being released to proceed under its own power. The performance is increased, but the prime reason for the high altitude launch is the added safety which 40,000 feet of altitude gives the pilot when he takes over under his own rocket power.

In considering a two-stage rocket aircraft which fulfills the requirements of this problem, several factors are of interest. In keeping with the basic philosophy of the project, i.e., repeatable, high altitude, manned flights, both the booster and final stages should be manned and recoverable. Such a design would have duplicate pilot and control needs, which, with the added complexity, would result in less efficiency and no doubt added cost. In other sections it is noted that the present configuration reaches the maximum altitudes from which safe re-entry and return to the earth can be accomplished, and that any gain in performance is unusable unless the satellite regime is attained. If the aircraft were designed with two-stages and still fulfilled the requirements of the study, the gain in performance would not be sufficient to achieve satellite status. It is therefore concluded that, with regards to power staging, the one-stage aircraft carried to altitude by a mother ship optimizes pilot safety, performance and overall simplicity.

MODEL 671

THREE-VIEW DIAGRAM

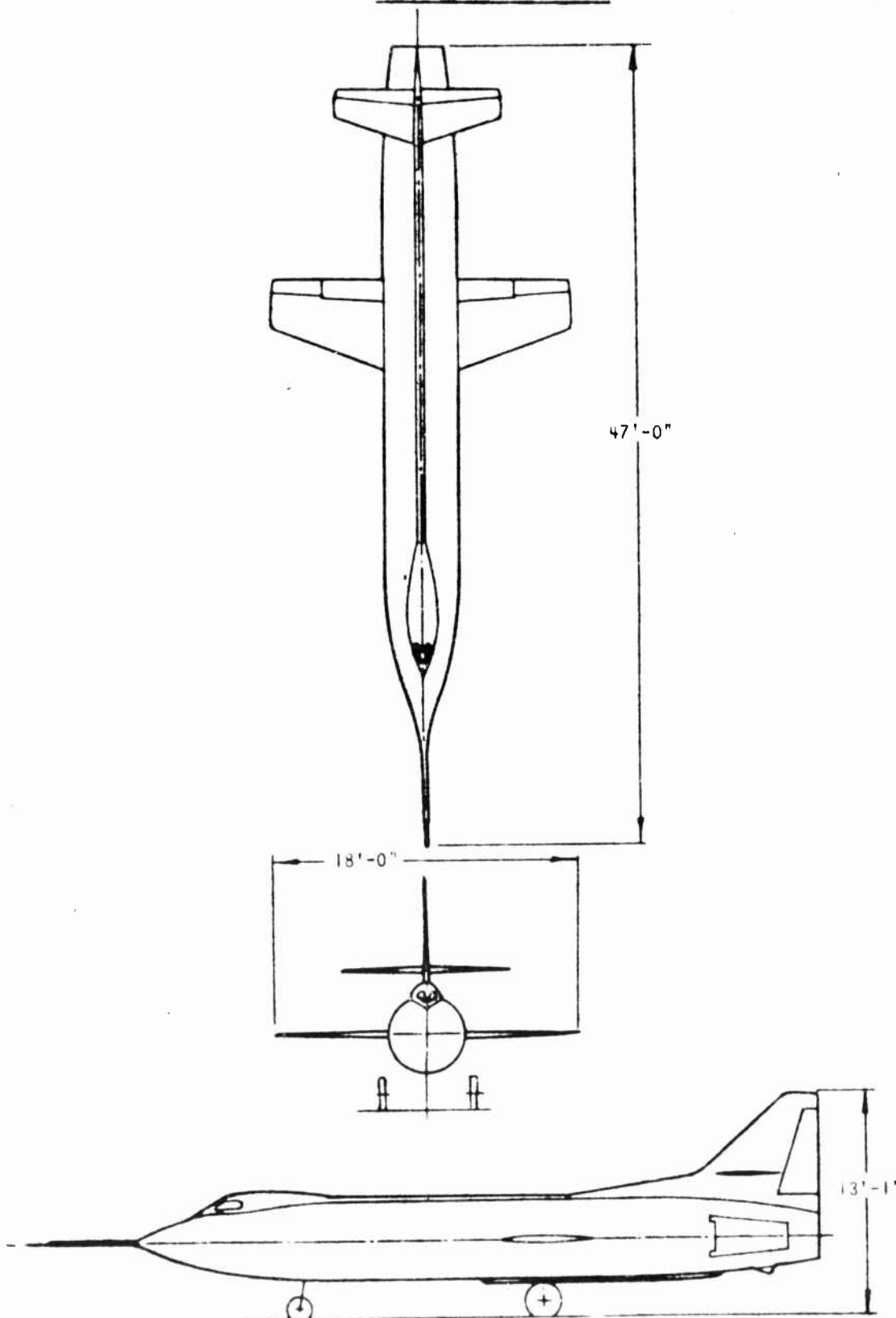


TABLE IMODEL 671DIMENSIONAL DATAWing

Span	ft.	18
Area	sq.ft.	81
Aspect Ratio		4.0
Taper Ratio		.5
Mean Aerodynamic Chord	ft.	4.7
Root Chord	ft.	6.0
Tip Chord	ft.	3.0
Airfoil Section		NACA 65A006

Ailerons

Span (one side)	ft.	3.3
Area Aft $\frac{1}{2}$	sq.ft.	2.3

Flaps

Span (one side)	ft.	3.3
Area Aft $\frac{1}{2}$	sq.ft.	3.8

Horizontal Tail

Span	ft.	9.8
Area	sq.ft.	24.3
Aspect Ratio		4.0
Airfoil Section		NACA 65A006

Elevator

Span (one side)	ft.	4.9
Area Aft $\frac{1}{2}$	sq.ft.	3.2

Vertical Tail

Span	ft.	6.2
Area	sq.ft.	32.4
Aspect Ratio		1.2
Airfoil Section		NACA 65A006

Rudder

Span	ft.	5.2
Area Aft $\frac{1}{2}$	sq.ft.	8.7

Fuselage

Length	ft.	41.3
Maximum Diameter	ft.	4.7

5.0 PERFORMANCE ANALYSIS

5.1 Calculation of Lift and Drag

The lift and drag for this preliminary study have been calculated by the use of standard methods of subsonic and supersonic analysis.

The lift curve slope of the wing was obtained from the results of DeYoung (Reference 1) corrected by a small empirical interference factor. The lift of the fuselage was based on the semi-empirical equations of Allen (Reference 2). The subsonic pressure and induced drag coefficients of all components of the airplane were calculated by standard semi-empirical methods.

The lift curve slope of the wing at supersonic speeds has been calculated by three-dimensional linearized theory. A correction has been made for the interference effect of the fuselage according to the method of Nielsen and Kaattari (Reference 3). The lift of the body itself was again obtained from the results of Allen (Reference 2).

There is no method available for the calculation of the supersonic, zero-lift, pressure drag of a finite wing with a laminar flow airfoil section. For the purposes of the present drag estimate, the following assumptions were made:

- (a) Three-dimensional planform effects are negligible for $M > 2.0$.
- (b) The two-dimensional section drag coefficient of a laminar flow airfoil can be approximated by that of a circular-arc section plus a proportionate part of the foredrag of a circular cylinder.

The three-dimensional effects on a wing are contained within the Mach cones from the root and tip. As the Mach number increases, the wing area affected by these cones decreases and the relative importance of the three-dimensional effects diminishes correspondingly. Three-dimensional theoretical results are available for wings with symmetrical double wedge sections, and these are compared in Figure 2 with calculations made by a simple "strip theory", ignoring the three-dimensional regions of the wing planform. It is apparent that the three-dimensional effects are negligible for Mach numbers greater than about 2, at least within the accuracy of the linear theory. It is assumed that the same conclusions apply to the use of the 65A006 airfoil section.

No theoretical methods have been devised for the calculation of the theoretical supersonic section drag coefficient of a blunt-nose airfoil. The pressure distribution can be approximated, however, by employing that for a circular cylinder over the nose until local sonic velocity is reached, and thereafter using a Prandtl-Mayer expansion. For the present analysis, it has been assumed that the drag of the 65A006 airfoil can be approximated by that of a 6%-thick symmetrical circular arc airfoil, with the blunt nose accounted for by applying the average of stagnation pressure and pressure at sonic velocity to 25% of the airfoil frontal area. This type of nose pressure distribution corresponds roughly to that at the front of a circular cylinder, and the 25% factor is obtained by comparison with test data (Reference 4). It has then been assumed arbitrarily that this method is applicable throughout the supersonic Mach number range. The resultant zero-lift pressure drag coefficient for the wing is shown in Figure 3.

The attitude drag of the wing was calculated from the standard formula $\Delta C_{D_w}/\alpha^2 = 1/C_{L\alpha}$.

The pressure drag of the fuselage was obtained from generalized curves based on the hypersonic similarity laws: the nose pressure drag from a correlation of characteristics solutions (Reference 5), and the boattail pressure drag from correlations of second-order solutions (Reference 6). Because of the relatively long cylindrical center section, interference between nose and boattail was neglected. The fuselage base drag was estimated from empirical correlation curves at the lower supersonic Mach numbers, and faired smoothly into values corresponding to a vacuum at the higher Mach numbers. The fuselage nose, boattail and base drag coefficients are plotted versus Mach number in Figure 4.

The fuselage attitude drag was computed by Allen's formula (Reference 2), just as for subsonic speeds.

The vertical and horizontal tail were treated in the same general way as was the wing, except that attitude drag was neglected. No allowance was made for drag due to trimming.

The drag coefficient of the dive brakes was based on an assumed normal flat plate drag coefficient of $C_{D90} = 1.8$. It was assumed that the brake drag coefficient was given by

$$\Delta C_{D_{br}} = \frac{S_{br}}{S_w} C_{D90} \sin \delta_{br}$$

For a brake area of 15 ft² and a maximum deflection angle of 60°, this gives a brake drag coefficient of $\Delta C_{D_{br}} = 0.30$.

The skin friction drag for the entire airplane was calculated using the usual flat plate formulas for laminar and turbulent flow. Forced transition was estimated to take place at the 20% station on the fuselage, and at the 50% station on the wing and tail surfaces. Natural transition was assumed to occur at a transition Reynolds number of 2.8×10^6 . The actual point of transition from laminar to turbulent flow was chosen at the more forward of these two locations.

Compressibility effects in laminar flow were accounted for by using a factor corresponding to the results of Crocco (Reference 7) and Van Driest (Reference 8). The corresponding correction for turbulent flow is difficult to determine. At the present time a number of different theories exist for the compressible turbulent boundary layer, all of which appear equally valid, but which lead to widely divergent results when extended to the higher Mach numbers (see for instance, Reference 9). The importance of the proper choice of a compressibility correction is emphasized by the fact that, for Mach numbers from 3 to 10, the uncorrected skin friction represents from 40 to 50% of the total zero-lift drag. For the present analysis, the theory of Van Driest (Reference 10) was chosen, which indicates relatively large decreases in turbulent skin friction with increasing Mach number, and therefore may be somewhat optimistic.

The airplane lift curves are plotted versus Mach number in carpet form in Figure 5. The transonic parts of the carpet have been faired appropriately. The total airplane drag coefficient is shown as a function of Mach number and lift coefficient in Figure 6 for a Reynolds number corresponding to a nominal altitude of 40,000 ft. Again, the transonic region has been faired.

5.2 Performance Methods in the Atmosphere

Because of the very transient nature of the entire flight, it is necessary to use a step-by-step method of solution of the equations of motion. The two equations which are necessary are

$$\frac{W}{g} \frac{dV}{dt} = T \cos \alpha - D - W \sin \gamma$$

$$\frac{W}{gV} \frac{d\gamma}{dt} = T \sin \alpha + L - W \cos \gamma$$

If the derivatives are replaced by finite increments, then the right sides of the equations can be considered constant for a small enough time increment, Δt , and the changes in velocity and flight path angle, ΔV and $\Delta \gamma$, calculated. The change in altitude is given by

$$\Delta h = \bar{V} \sin \bar{\gamma} \Delta t$$

Thus, a complete history of the flight path within the atmosphere may be calculated. This method was used for the power-on ascent and for the pullout.

5.3 Performance Methods in Space

The flight path in space could be calculated with sufficient accuracy from the equations presented above, but because of the large time spent in flight outside the atmosphere, it is much simpler to integrate the equations to obtain the usual ballistic space trajectory equations. This gives the relations:

$$V^2 = V_0^2 - (2 g V_0 \sin \gamma_0) t + g^2 t^2$$

$$V \cos \gamma = V_0 \cos \gamma_0$$

$$h = h_0 + (V_0 \sin \gamma_0) t - \frac{1}{2} g t^2$$

from which the velocity, flight path angle, and altitude can be calculated at any time for the given initial conditions, V_0 , γ_0 , h_0 . In general, it has been assumed that the ballistic trajectory equations were applicable for $h > 300,000$ ft.

From these equations, the peak altitude is given as

$$h_p = h_0 + V_0^2 \sin^2 \gamma_0 / 2g$$

with

$$V_p = V_0 \cos \gamma_0$$

$$t_p = t_0 + V_0 \sin \gamma_0 / g$$

The time to return again to h_0 is, obviously,

$$t_m = t_0 + 2V_0 \sin \gamma_0 / g$$

5.4 Discussion of Performance Parameters

Some discussion is desirable of the parameters affecting the performance of this airplane, because many of the problems are entirely different from those of the more conventional airplane or missile.

The function of drag in the overall performance must be reconsidered. The effect of drag is practically negligible in the power-on ascending phase of flight (for a high altitude air launch), because of the very large thrust to weight ratio. Throughout the vacuum trajectory the aerodynamic shape of the airplane is completely unimportant. During the descending phase of the flight, a large drag is very beneficial in aiding in the pullout, and the highest possible drag is desired within the limits of the pilot and the structure. In fact, during the pullout it has been assumed that drag brakes would be extended in order to decelerate as soon as possible. However, because of excessive decelerative forces acting upon the pilot, it is necessary to gradually retract the brakes as denser air is entered, until they are fully retracted in the later stages of the pullout.

For a given propulsion unit (i.e., fixed thrust and fuel consumption), the overall performance of the present design is much more dependent upon the ratio of fuel weight to gross weight than it is upon the minimum drag or the optimum lift-drag ratio. Even though the fuel is expended in approximately the first 75 seconds of flight (a relatively small fraction of the total flight time), the ultimate performance as measured by the maximum altitude is affected to a great extent by small changes in the fuel to gross weight ratio. As an example, an increase in fuel weight/gross weight from 0.65 to 0.70 results in an increase in peak altitude of about 35% for a typical vertical flight trajectory, other parameters remaining constant. In addition, the pullout conditions are noticeably altered by the same relative changes, and, as will be discussed later, the pullout presents the limiting factors which determine the optimum performance. Therefore it is all important, in the present design, that the structural and equipment weight be kept to an absolute minimum in order to achieve satisfactory performance.

Early in the program the relative merits of a ground take-off versus an air launch were studied. Although it was considered desirable

from an operational standpoint to use an unassisted ground take-off, it was immediately apparent that the performance penalty was too large to make this feasible. An increase in launch altitude from sea level to 40,000 feet resulted in a 200,000 foot increment in maximum altitude for a typical flight path. The additional benefits of higher altitude launch disappear rapidly above 40,000 feet, since most of the improvement is due to the decreasing air density, although the non-zero launch speed is a contributing factor.

5.5 Flight Path Calculations

Using the methods discussed above, a series of flight paths has been computed for the airplane configuration described in Section 4.0, except that the full and empty gross weights used in the performance analysis were 20,550 lbs. and 5550 lbs., respectively. The effect of the weight change will be discussed later. The basic plan of the flight path is given in detail in the following steps.

1. Air launch the airplane at an altitude of 40,000 feet and a true air speed of 495 knots ($M = 0.75$). These are considered to be practical conditions for a suitable carrier airplane.
2. Apply full thrust immediately after launch. The engine parameters used in all of the analyses are:
 - a. Thrust $T = 50,000$ lbs.
 - b. Specific impulse $I = 250$ sec.
 - c. Fuel flow $(T/I) = 200$ lb./sec.
3. Pull-up at maximum lift coefficient until a given flight path angle is reached. The maximum lift coefficient was programmed with $C_{L_{max}} = 0.4$ from $M = 0.75$ to $M = 1.0$, and $C_{L_{max}} = 1.0$ supersonically.
4. Fly a power-on, zero-lift trajectory until the fuel is exhausted. This occurs at 75 seconds.
5. Follow a vacuum ballistic trajectory until the atmosphere is re-entered in the descent (roughly 300,000 ft.).
6. Pullout to level flight. This is accomplished by the following operations.
 - a. The lift is increased to a maximum, subject to the following limitations: (1) a maximum lift coefficient

of 1.0; (ii) a maximum angle of attack of 30° ;
(iii) a maximum normal acceleration of $6g$.

- b. The dive brakes are opened fully until a longitudinal deceleration of $4g$ is attained. The brakes are then gradually retracted so as to maintain this deceleration, until they are fully retracted.

This flight plan has been followed for four different flight paths, which have been identified arbitrarily by the approximate flight path angle at burnout as the 29° , 38° , 52° and 84° flights. Time histories of the altitude, velocity, resultant acceleration, and flight path are plotted in Figures 7, 8, 9, and 10. Some of the important information is summarized in Table II.

The complete flight path calculations discussed above have been made for a take-off weight of 20,550 lbs., with 15,000 lbs. of propellant. The final weight estimate for the airplane has increased the take-off weight to 22,200 lbs., with the same propellant weight. The basic 38° flight path has been recalculated for the new weight, also including the predicted variation of thrust with altitude (see Figure 26). The increase in thrust more than compensates for the increase in weight, and the altered performance is slightly better than that shown here.

No attempt has been made in the present study to determine an absolute optimum flight path, because of the large number of variables involved. Certain procedures have been used which generally are optimizing, such as flying a boost-glide path (which burns all of the fuel in the shortest possible time) to provide the most efficient use of energy, and climbing at as near a vertical flight angle as possible in order to convert all of the energy into altitude. Unfortunately, the restricting conditions of the pull-out affect the optimum values to a great extent, so that it is not certain that these procedures are really optimum for the present configuration. It seems probable that a systematic investigation of flight path parameters would permit an increase in performance, but indications are that the improvements would be slight.

5.6 The Re-entry Problem

The problems connected with re-entry into the atmosphere require very careful and detailed consideration, since above all others, they represent the factors which determine and limit the overall performance. In particular, the following items are critical.

1. Altitude of pullout
2. Acceleration and deceleration
3. Structural and environmental temperatures
4. Time duration

The effects of the first two problems have been investigated by a short generalized study. Following the pullout pattern described in Section 5.5, a series of pullouts was calculated for arbitrarily chosen initial flight conditions in descent at 300,000 feet altitude. By cross-plotting the pullout altitudes and resultant accelerations it was possible to establish curves of the flight conditions at 300,000 feet which are necessary to attain a given pullout altitude or a given net acceleration. Several such curves are shown in Figure 11. Superimposed on the same figure is a curve showing the conditions at 300,000 feet corresponding to a maximum altitude of 700,000 feet. It is apparent immediately that the choice of flight path is quite limited if it is desired to attain a maximum altitude greater than 700,000 feet and a pullout altitude higher than 30,000 feet. A curve representing the capabilities of the present airplane-engine-fuel combination is also indicated on the figure; the maximum peak altitude attainable with this airplane consistent with the 30,000 foot pullout limit is 770,000 feet.

The normal acceleration (lifting force) in the pullout has been arbitrarily limited to 6g, and the dive brakes have been retracted so as to maintain a drag force corresponding to a 4g deceleration, so that the majority of the pullouts have a net acceleration of no more than the resultant of these (7.2g). However, for the more critical flight conditions the drag of the airplane with fully-retracted brakes exceeds a 4g load, and the resultant acceleration increases. The maximum resultant accelerations are indicated approximately on the figure.

It is interesting to note that the problems of pullout altitude and limiting accelerations, which are intimately related, are traceable directly to the single limiting factor of the presence of a human pilot. Increases in the peak altitude are attainable only by increasing the speed and/or flight path angle at 300,000 feet. But it would still be possible to pullout at a safe altitude by simply increasing the lift, increasing the drag, or both. Therefore the peak altitude which is attainable is fundamentally limited by the probable maximum resultant acceleration which the pilot can withstand in the pullout. Under these conditions the peak performance is practically independent of the aerodynamic configuration.

The aerodynamic heating effects which cause the high boundary layer and skin temperatures are also dependant upon the production of drag, but with one important difference. Whereas deceleration is caused by the total of all drag forces acting on the airplane, aerodynamic heating is related only to the skin friction drag. Therefore, it will be beneficial to dissipate as much energy as possible in the form of pressure and wake drag, for a given deceleration. In other words, for a given decrease in velocity, the aerodynamic heating will be kept at a minimum by producing as much drag as possible by drag brakes, high angle of attack, or a blunt configuration, disregarding local effects such as stagnation points, etc.

The fourth critical item is the length of time of the pullout. Since the entire operation, from the time of entry into the atmosphere until level flight is attained, is accomplished in less than two minutes, the timing of each phase of the maneuver must be extremely accurate. Considering the high accelerations and high temperatures to which the pilot will be subjected during this period, as well as the possible consequences of deviations from the proper procedure, it seems probable that some type of automatic control may be necessary or that the time for pullout must be increased in some manner. On the other hand, the time spent in the pullout is marginal with respect to subjecting the pilot to high accelerations for long periods; from this standpoint, it would be desirable to decrease the length of time of the pullout.

One possible overall solution to the critical problems of the pullout lies in the use of braking thrust during descent. If a certain amount of the total fuel is not burned during the initial boost period, this can be used to provide a braking thrust during the descent, either by a mechanical reverse thrust device or by a tail first descent. For a given ballistic flight path angle, the peak altitude will be reduced because of the incomplete burning during boost, but with sufficient braking thrust the flight path angle can be increased. It seems certain that definite optimum values exist for the percentage of fuel and flight path angle to give a maximum peak altitude for given pullout conditions; however, no attempt has been made at this time to establish these optimums. In order to indicate the trend, one flight path has been calculated in which the ballistic flight path angle has been increased to approximately 90° and roughly 10% of the fuel has been saved for use in braking (Figure 12). It will be noticed that a peak altitude is reached which is comparable to the best of the standard flights, while the pullout is satisfactory with regard to altitude. The load factor remains within the prescribed limits, and the boundary layer temperature is reduced to below 2000° F. According to Figure 11, the maximum permissible falling

velocity for safe recovery from a vertical dive is 3500 ft/sec at 300,000 ft. altitude. Calculations have been made of the vertical heights attainable as a function of the percent of propellant carried. The results are plotted in Figure 13. At a value of $\gamma = .573$ the natural falling velocity at 300,000 ft. is 3500 ft/sec. Below this value the natural falling velocity at 300,000 ft. would be less than 3500 fps. Above this value of γ the natural falling velocity would exceed 3500 ft/sec. and some propellant must be assigned to the braking operation. The chart shows the fraction of the total that must be assigned for braking, and indicates the loss in altitude necessitated by the braking operation to assure a safe descent.

While this method of improving the flight conditions appears to be promising, it involves a number of serious additional problems. There may be control difficulties in a tail-first descent, and additional control problems in returning to a normal flight attitude after the final burnout. The relighting is extremely critical with either method of braking with thrust, since failure to restart or even a delay in restarting would endanger the pull-out.

The very high decelerations that occur during the descent without reverse thrust are a basic phenomenon that cannot be avoided unless a near-satellite path is approached. The basic difficulty can be brought out in a qualitative manner quite simply by consideration of a vertical flight. During powered ascent, acceleration can continue even after the rocket is above the atmosphere. But, because the atmosphere is used as the means for deceleration, deceleration can occur only within the atmosphere. For simplicity consider a motion having constant acceleration. If burnout occurs at a height of 4 atmospheric thicknesses, where the atmosphere is arbitrarily regarded as 100,000 ft. thick, then $V^2 = 2ah = 2a(4 \times 100,000)$. But this same velocity must be dissipated during 100,000 ft. of descent, so the deceleration must be four times as great as the acceleration. Greater performance will only magnify the deceleration problem because all decelerations must occur in a fixed distance. The only other solution occurs if performance becomes so great that the aircraft enters the atmosphere tangentially. Then the aircraft can use a very long distance in which to decelerate. It follows that there appears to be a "forbidden region" of flight from which safe descent becomes impossible. Figure 14 is a plot of $h = \frac{V^2}{2a}$ included to indicate the order of magnitude of distances required to accelerate to various velocities at various rates of acceleration. It shows clearly that very high speeds can be attained only by using long distances for acceleration, or else by using very high acceleration rates. The chart of course is equally applicable for decelerations.

5.7 Landing

As in the case of other research projects such as the Model D-558, a relatively high landing speed is accepted in order to improve the high speed and altitude performance for which this project is intended. The stalling speed at landing weight is 154 knots, which results in an approach speed of approximately 185 knots. This value could be decreased by utilizing high-lift leading-edge devices or by increasing the wing area. However, the increased weight and/or the resulting complications in the leading-edge cooling system appear to make these changes undesirable - particularly since other research aircraft have demonstrated that satisfactory operation can be obtained with landing speeds of this magnitude.

MODEL 671

WING PRESSURE DRAG COEFFICIENT COMPARISON OF THEORIES

DOUBLE WEDGE SECTION

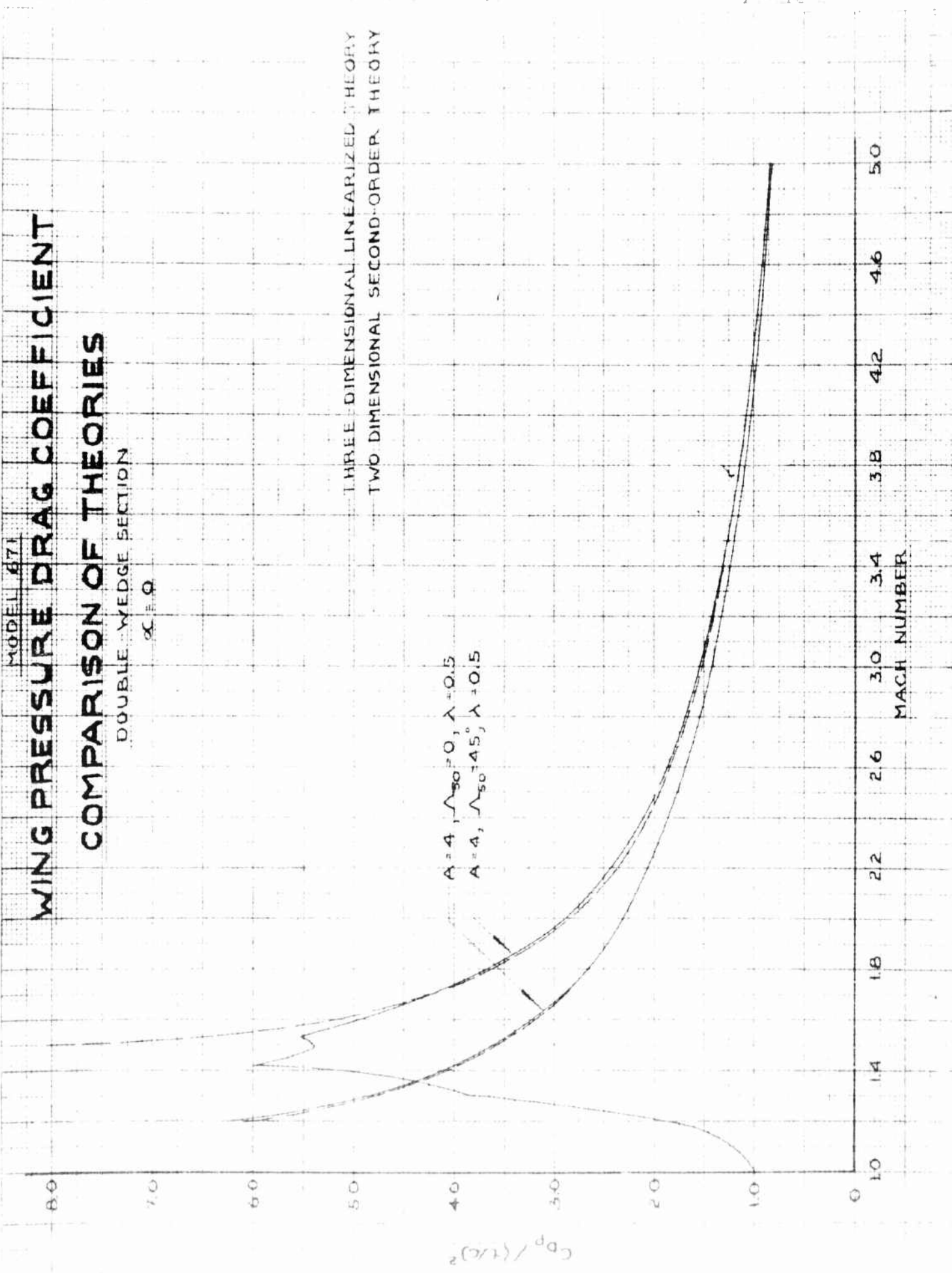
$\alpha = 0$

THREE DIMENSIONAL LINEARIZED THEORY
TWO DIMENSIONAL SECOND-ORDER THEORY

$A=4, \Lambda_{50}=0, \lambda=0.5$
 $A=4, \Lambda_{50}=45^\circ, \lambda=0.5$

$C_{Dp}/(\gamma M_\infty^2)$

MACH NUMBER



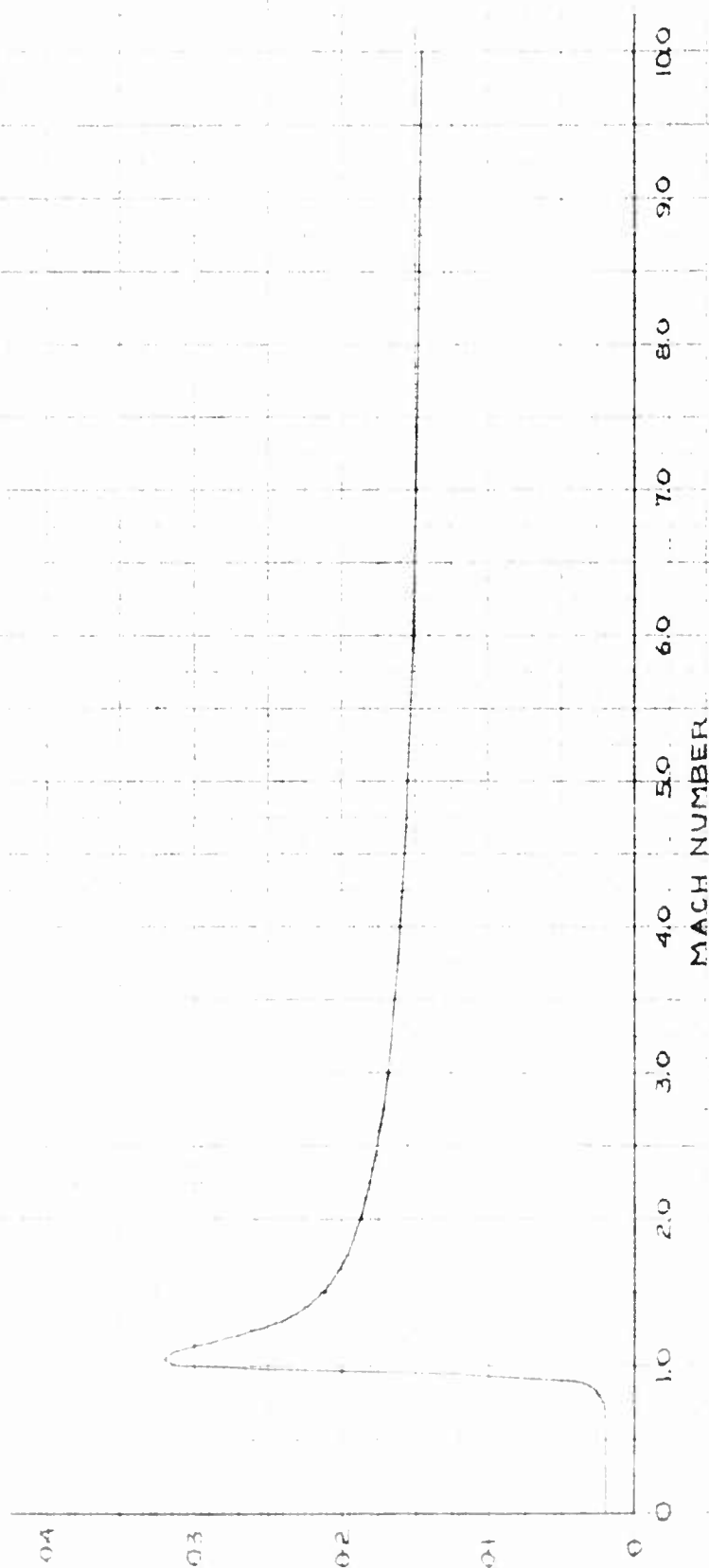
FORM 10-1, AG-1, 1-1-41

MODEL 671

WING PRESSURE DRAG COEFFICIENT

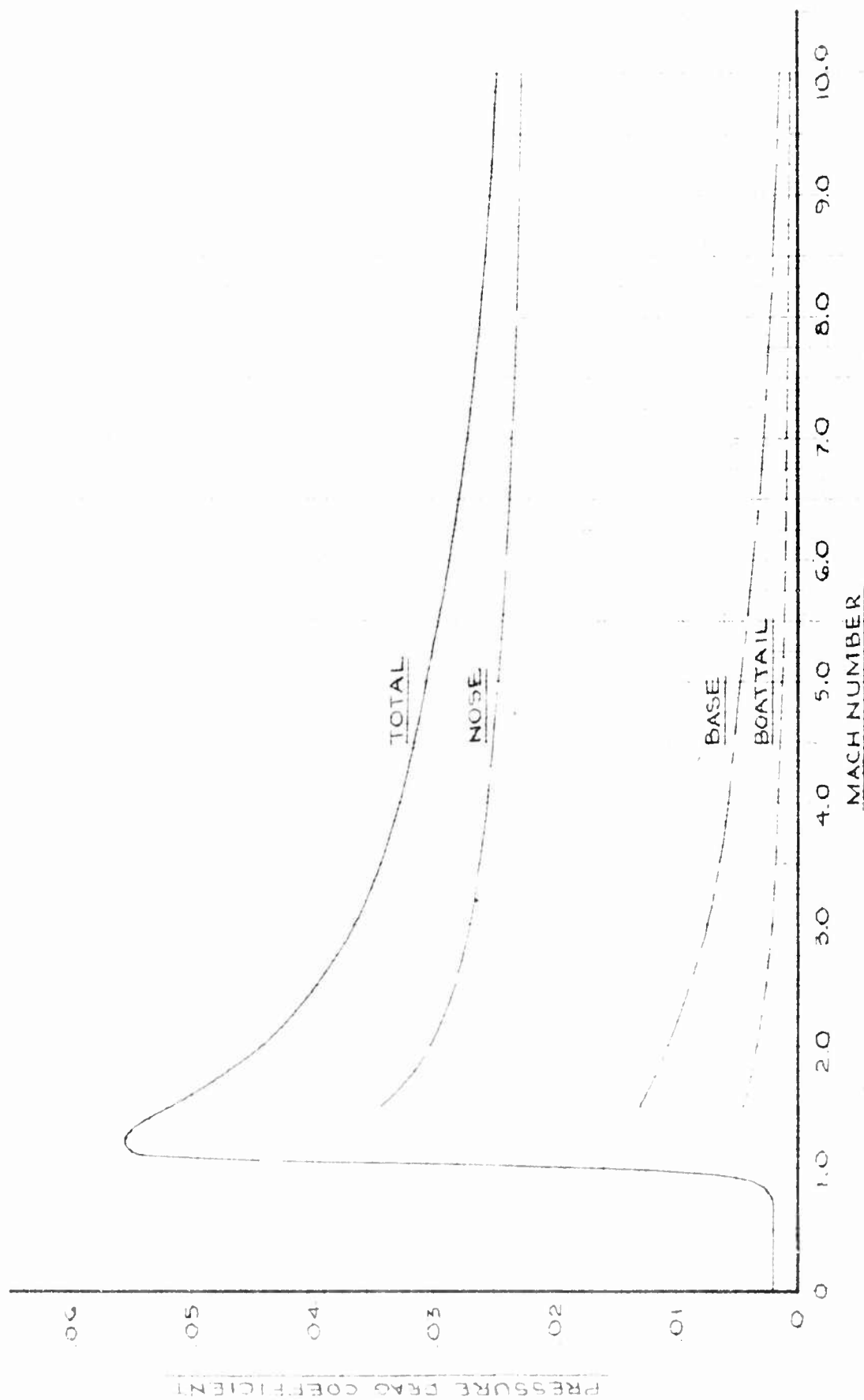
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WING PRESSURE DRAG COEFFICIENT C_{Dp}



MODEL 671

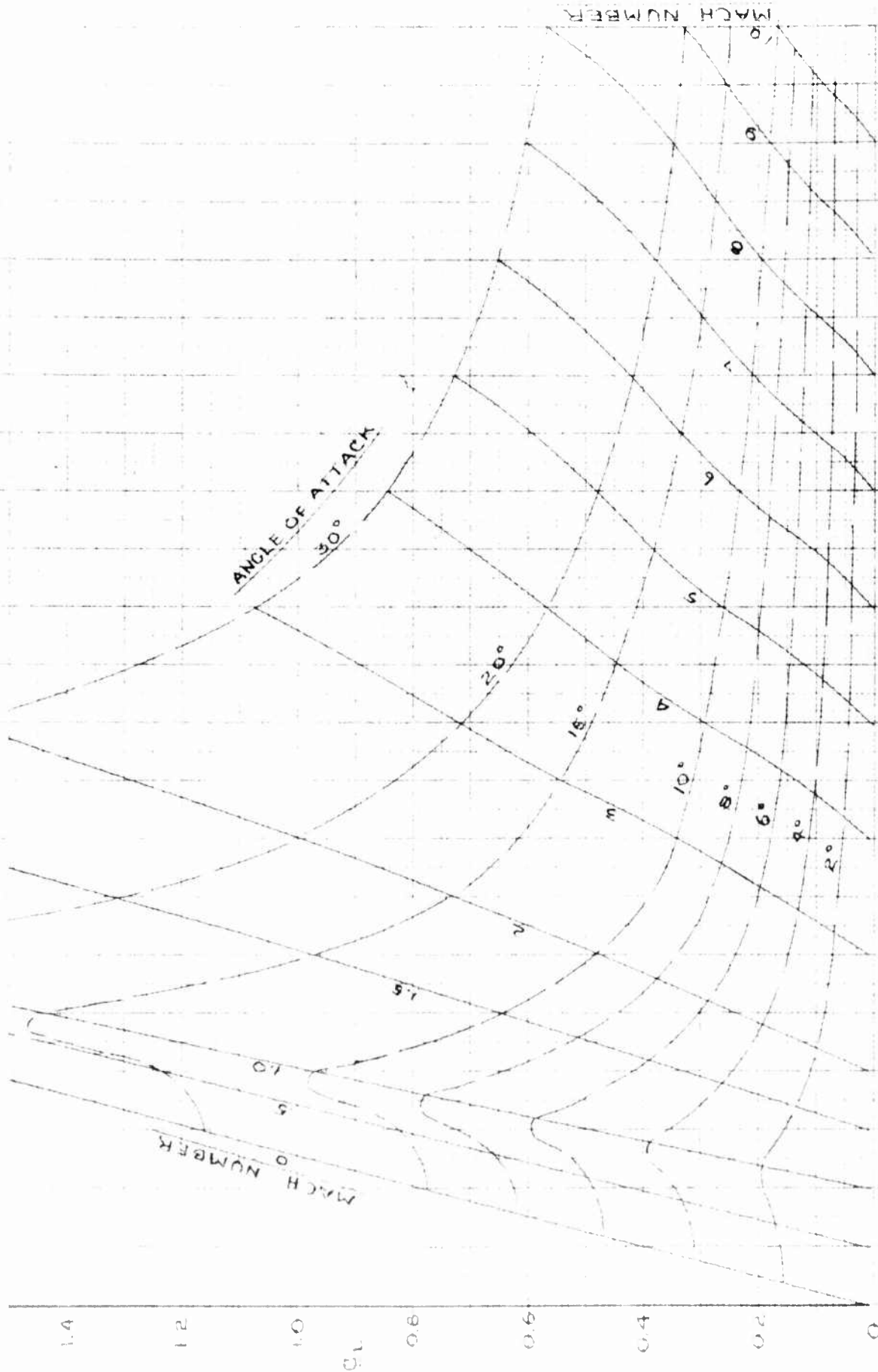
FUSELAGE PRESSURE DRAG COEFFICIENT



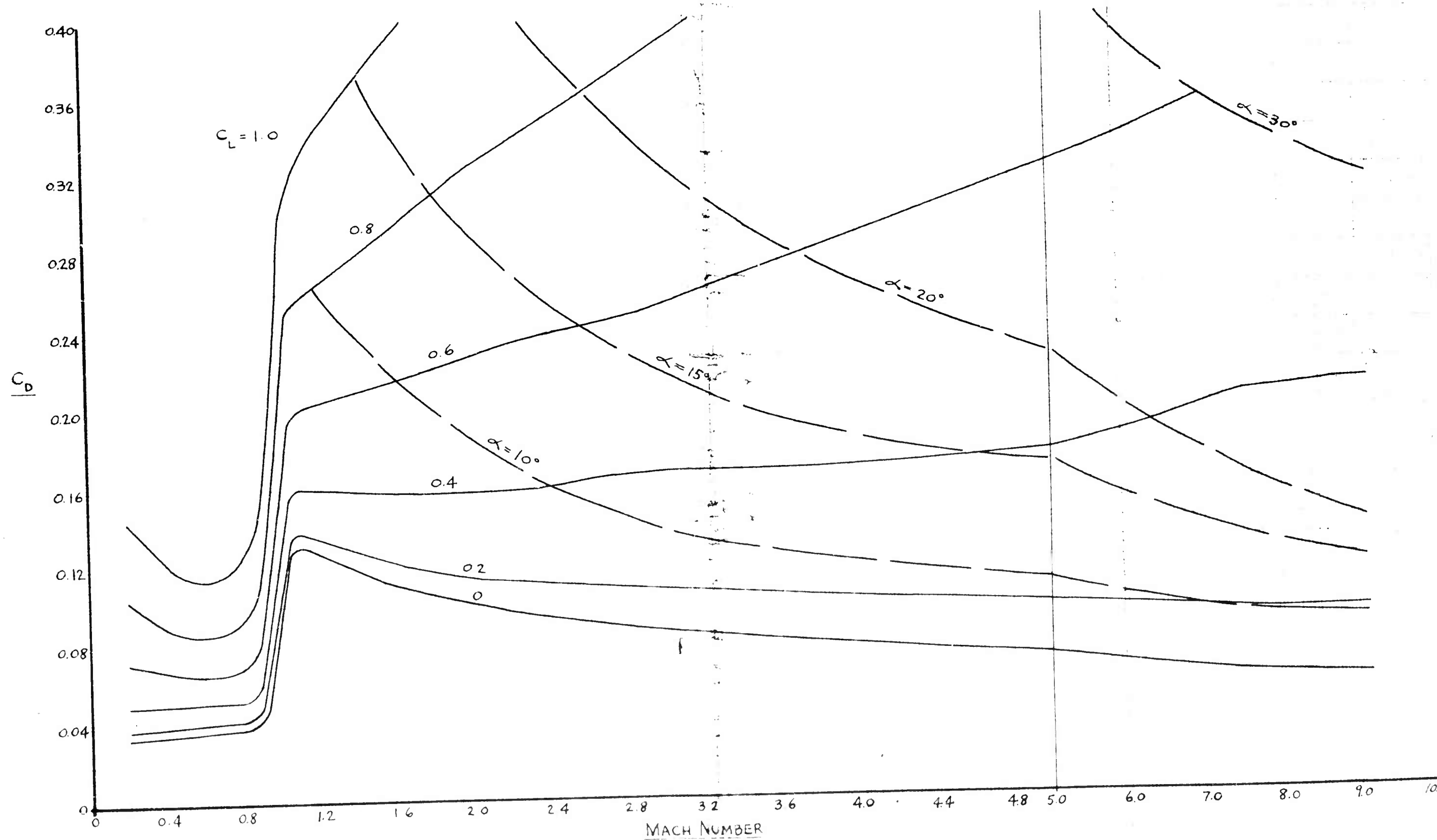
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MODEL 671

AIRPLANE LIFT COEFFICIENT CARPET

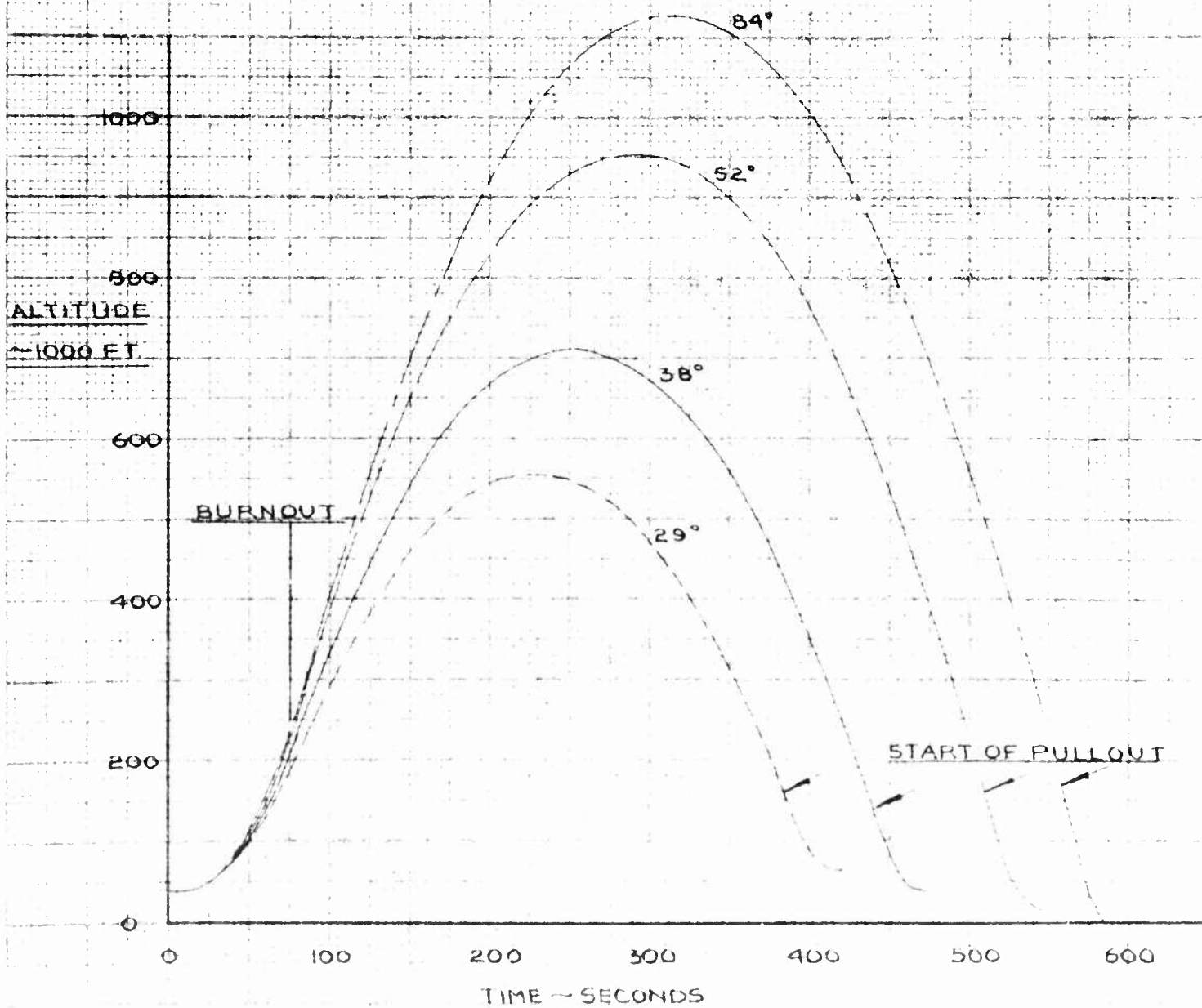


MODEL 671 AIRPLANE DRAG COEFFICIENT



MODEL 671

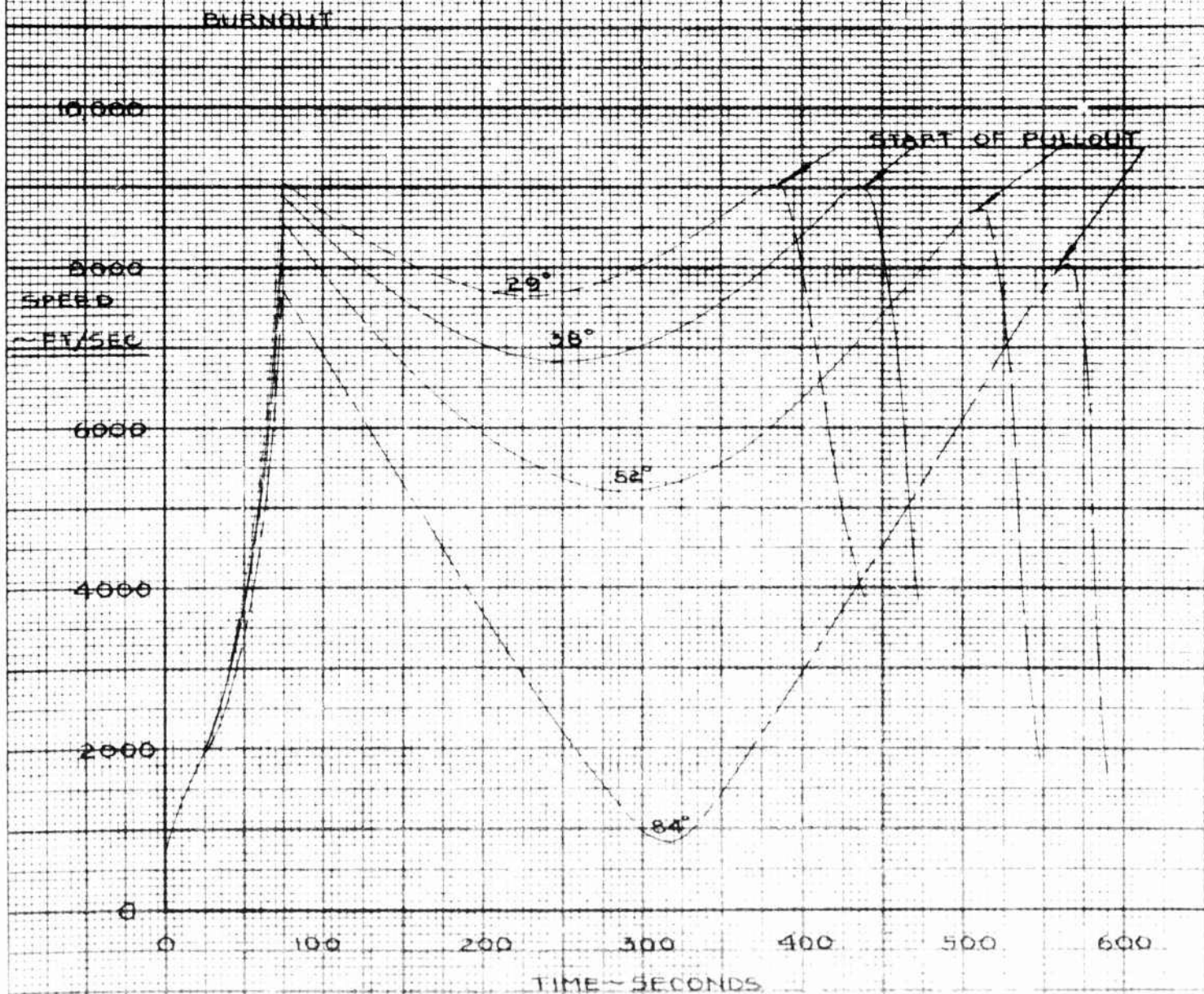
EFFECT OF TRAJECTORY ANGLE ON FLIGHT PATH ALTITUDE VS TIME

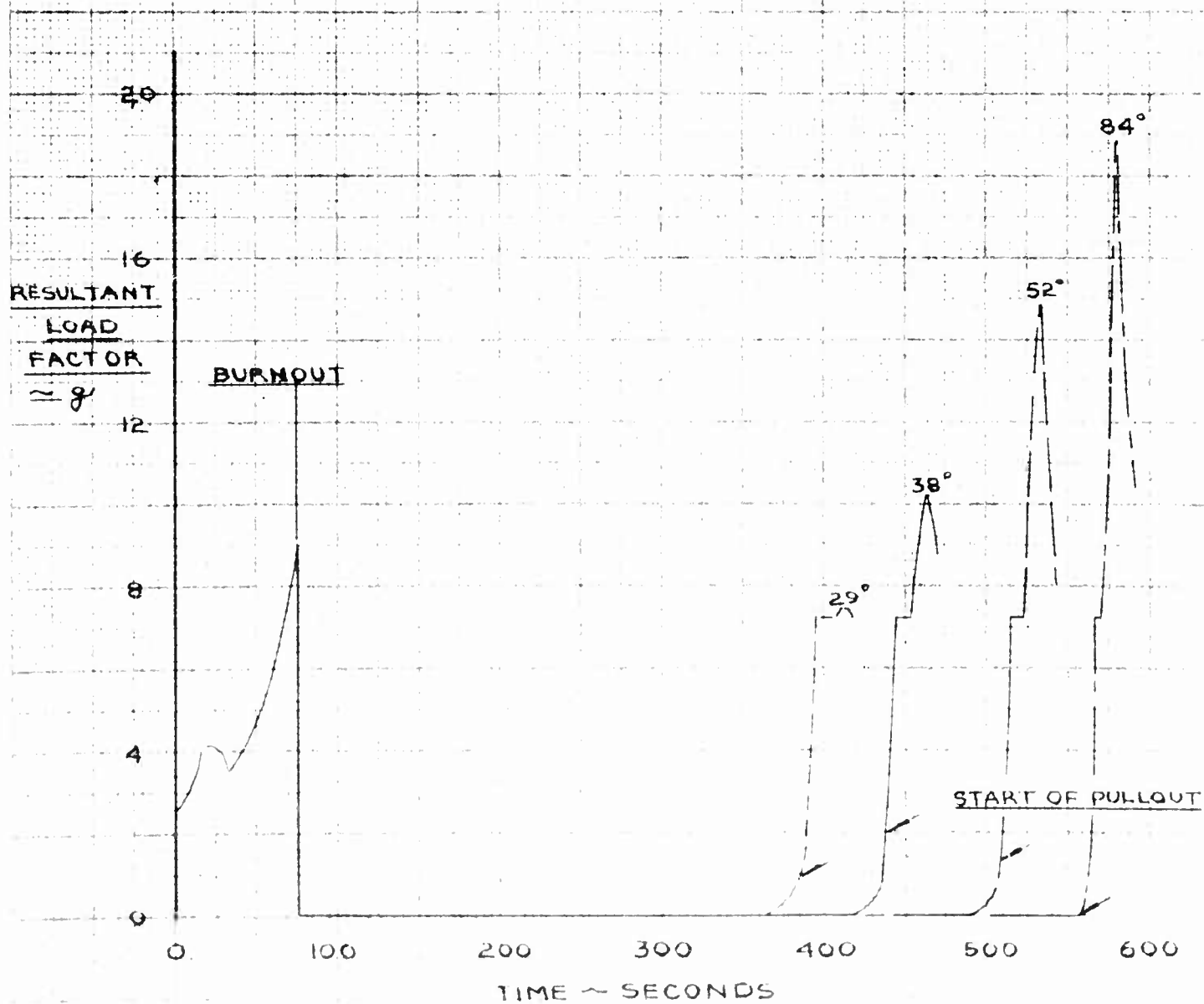


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MODEL 671

EFFECT OF TRAJECTORY ANGLE ON FLIGHT PATH SPEED vs TIME



MODEL 671EFFECT OF TRAJECTORY ANGLE
ON FLIGHT PATH
RESULTANT LOAD FACTOR VS TIME

MODEL 67

SPACE TRAJECTORY

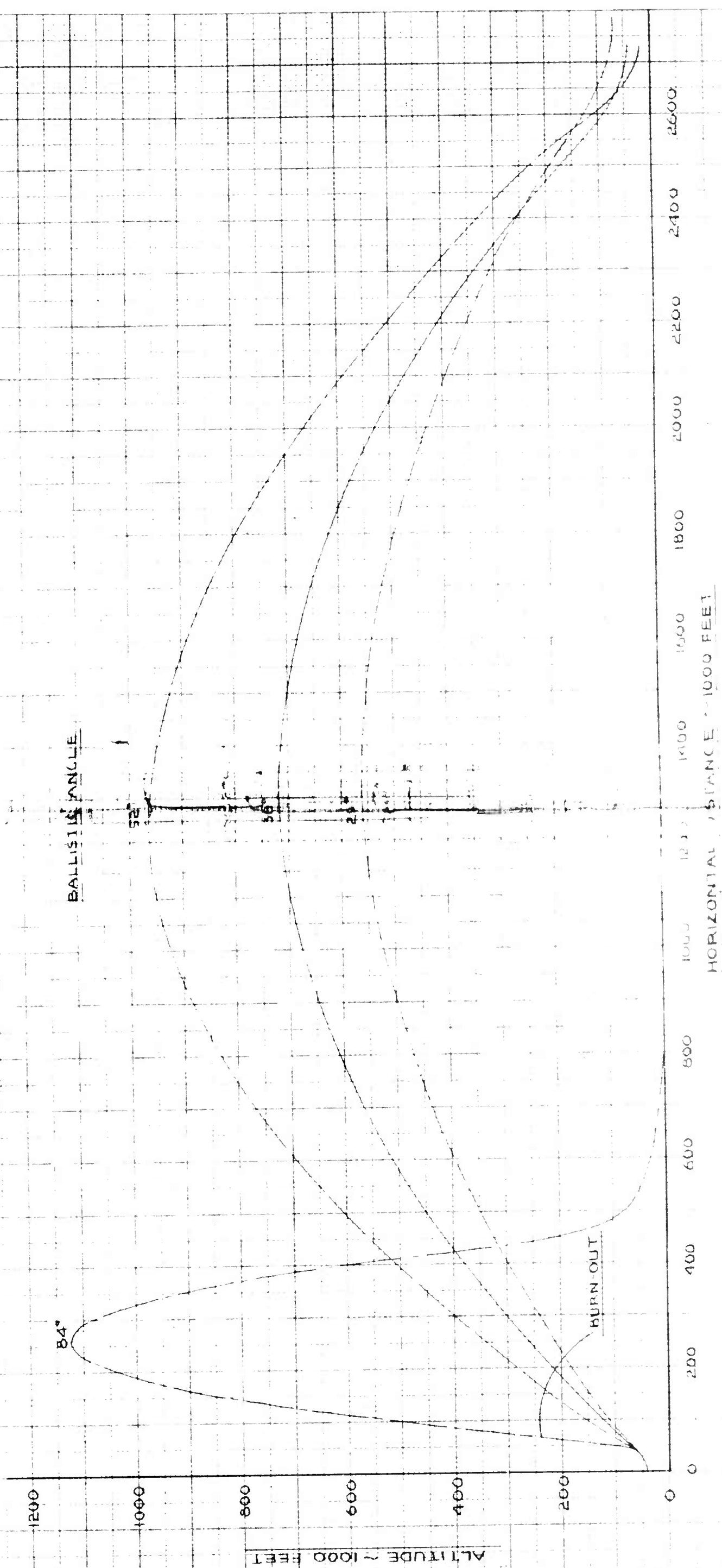


TABLE II
FLIGHT TRAJECTORY SUMMARY

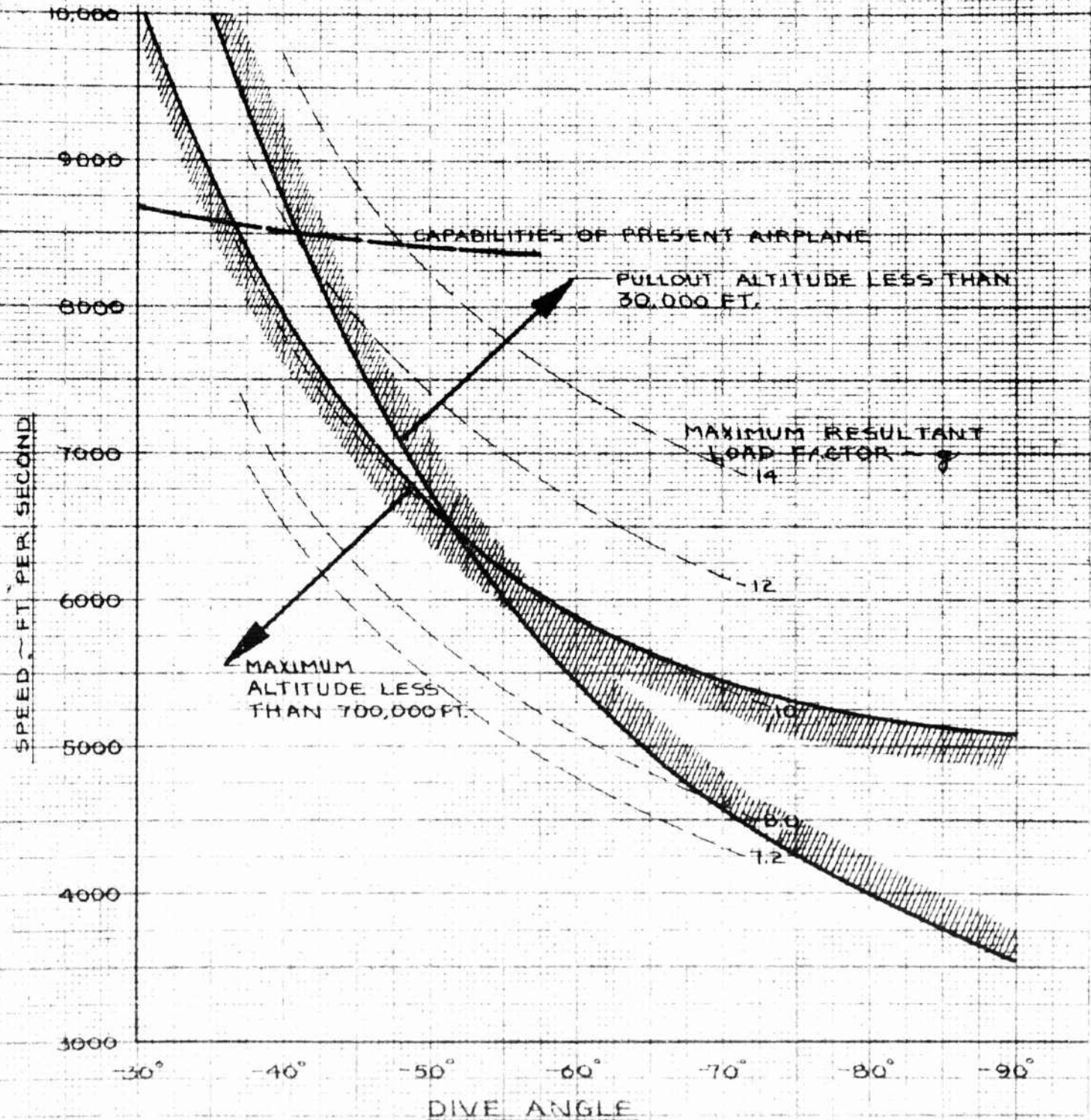
	<u>29°</u>	<u>38°</u>	<u>52°</u>	<u>84°</u>
Peak Altitude (ft)	555,000	710,000	953,000	1,126,000
Velocity at Peak (ft/sec)	7,650	6,830	5,200	830
Velocity at Burnout (ft/sec)	9,030	8,870	8,550	7,690
Angle at Burnout (deg)	32.3	39.7	52.6	84.1
Pullout Altitude (ft)	61,800	39,100	17,100	---
Velocity at Pullout (ft/sec)	5,610	3,800	2,000	---
Maximum Resultant <i>g</i>	7.4	10.2	14.8	18.9

DOUGLAS AIRCRAFT COMPANY, INC. 1. SECOND STREET, PALM BEACH, FLORIDA

MODEL 671

RESTRICTIONS ON FLIGHT CONDITIONS **AT ENTRY INTO ATMOSPHERE**

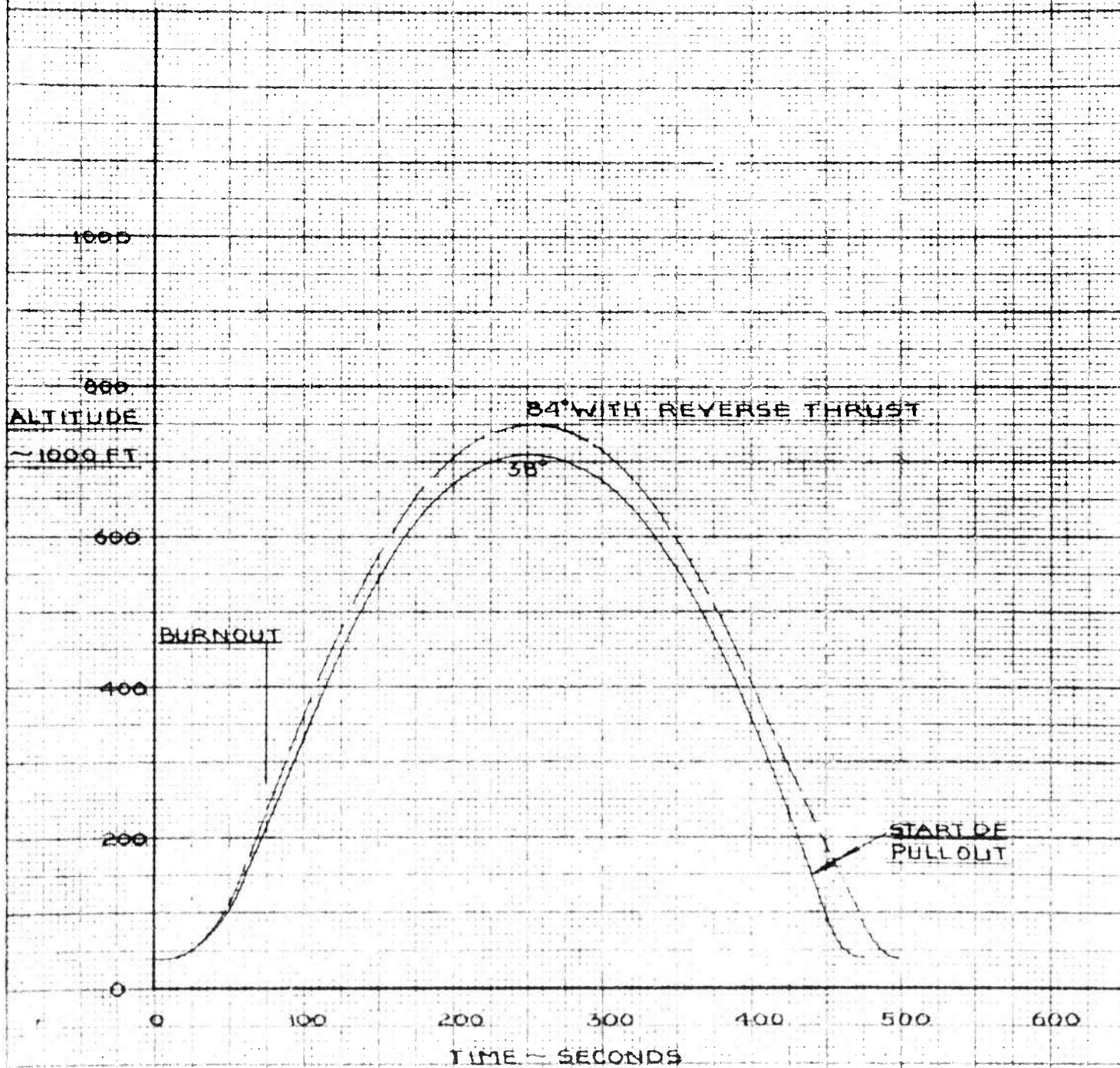
(300,000 FEET ALTITUDE)



DOUGLAS AIRCRAFT COMPANY, INC. 10000 WILSON AVENUE, BOSTON, MASSACHUSETTS 02116

MODEL 671

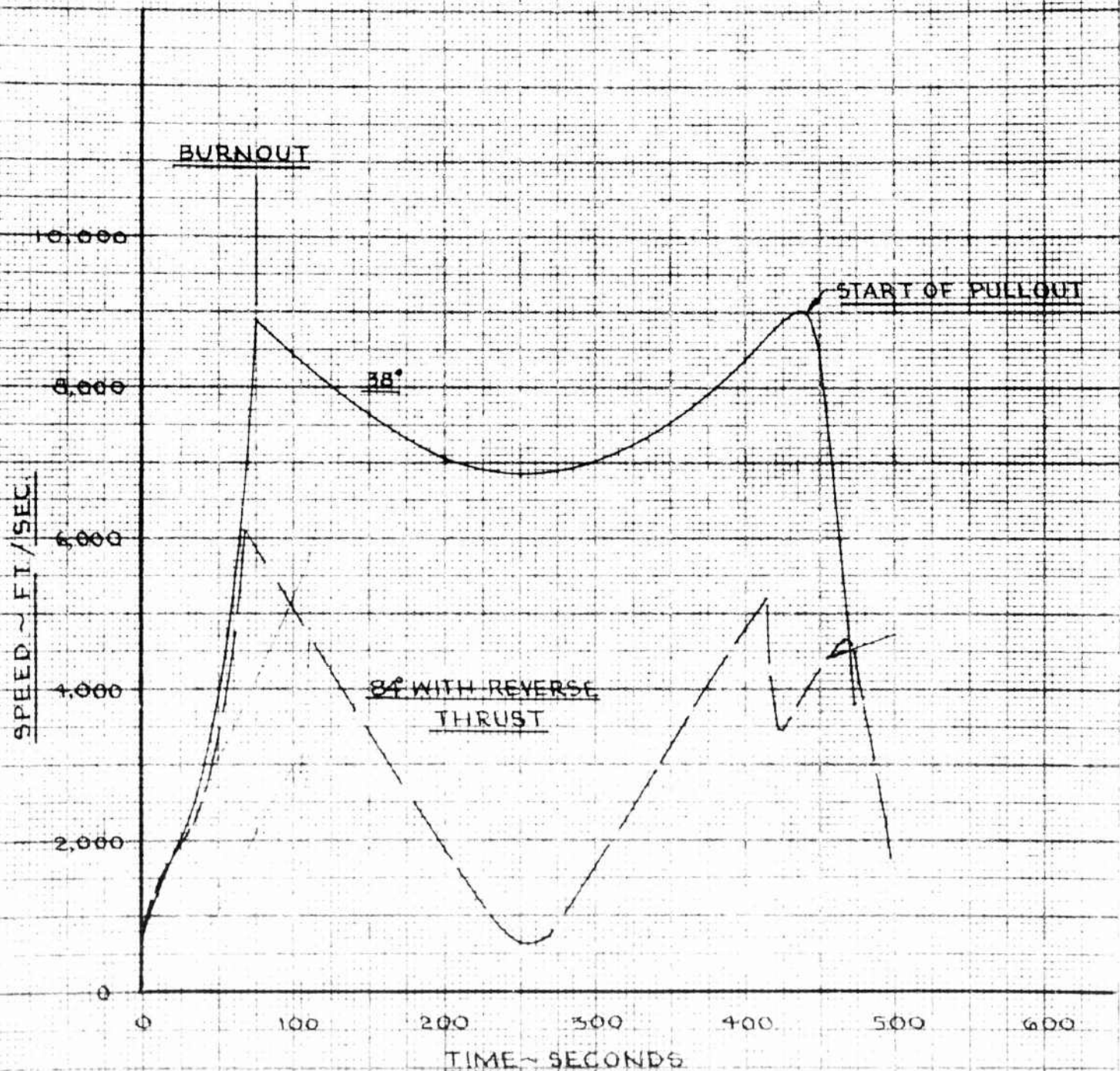
EFFECT OF REVERSE THRUST ON FLIGHT PATH ALTITUDE vs TIME



DOUGLAS AIRCRAFT COMPANY, INC. EL SEGUNDO PLANT EL SEGUNDO, CALIFORNIA

MODEL 671

EFFECT OF REVERSE THRUST ON FLIGHT PATH
SPEED vs TIME



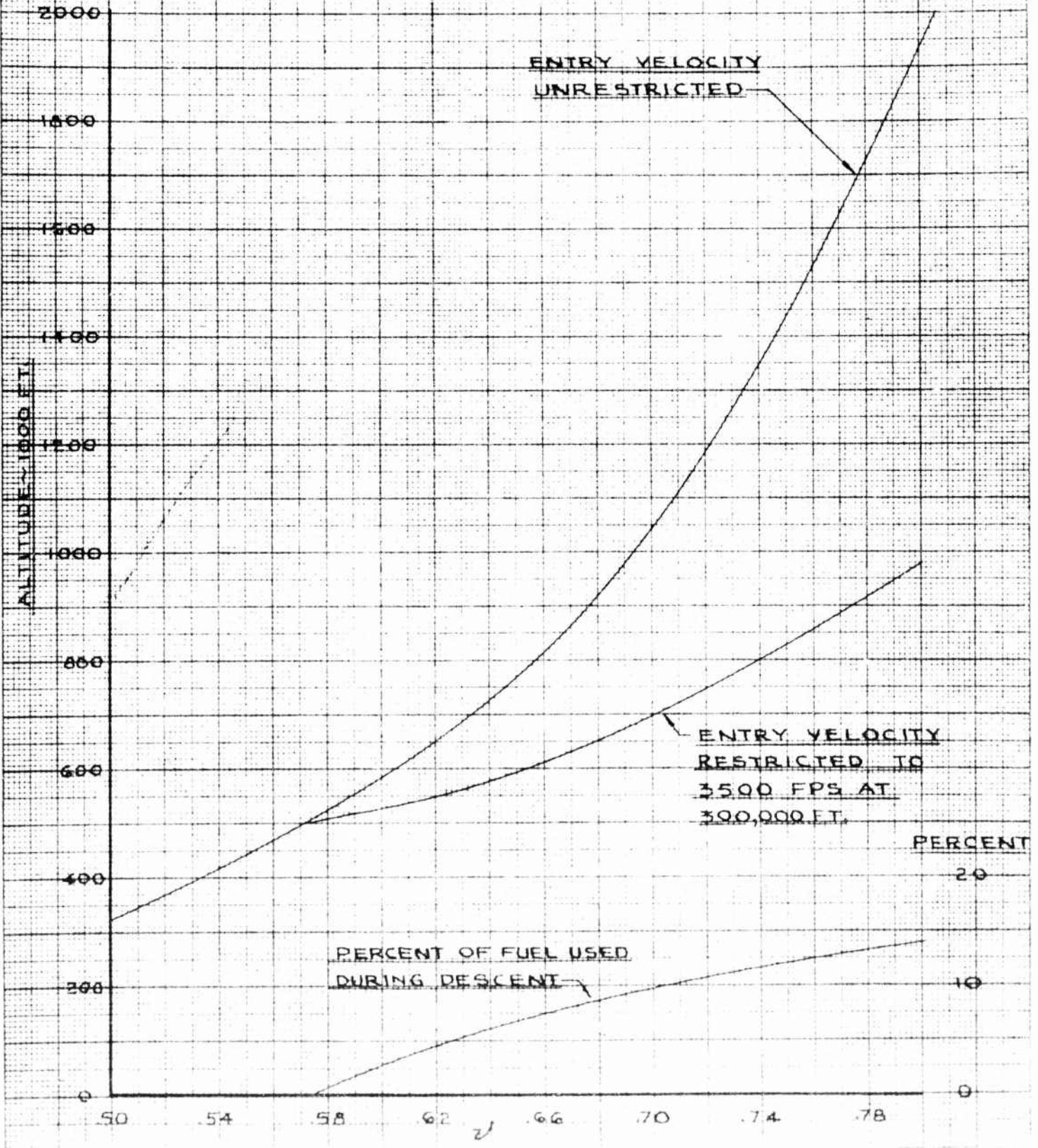
DOUGLAS AIRCRAFT COMPANY, INC. EL PASO, TEXAS

MODEL 671

EFFECT OF THE USE OF REVERSE THRUST DURING DESCENT

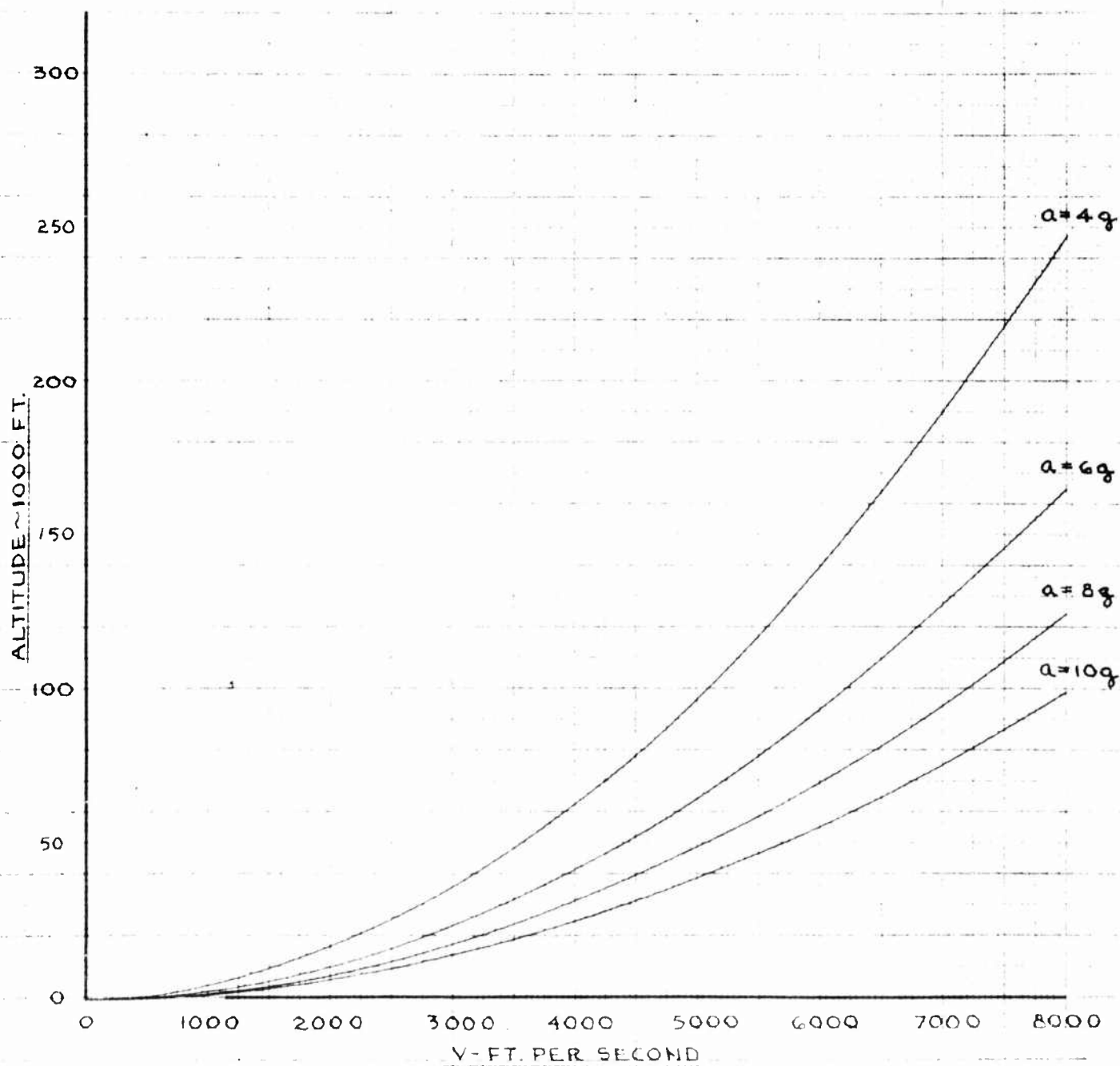
SPECIFIC IMPULSE = 250 SEC.

VERTICAL FLIGHT IN VACUUM



MODEL G71

ALTITUDE vs. VELOCITY FOR VARIOUS RATES OF ACCELERATION



6.0 STABILITY AND CONTROL

6.1 Flight Within the Atmosphere

The portion of the flight within the atmosphere can be broken down into the following regimes for the purpose of stability and control discussion:

1. Launching
2. Recovery to level flight
3. Landing

Since requirements for satisfactory launching and landing are known or can be adequately predicted on the basis of past experience, it is considered necessary to discuss these phases only very briefly at the present time.

6.1.1 Launching Stability

The aircraft will be air-launched at maximum practical altitude and airspeed and will accelerate rapidly to supersonic speeds. On the basis of experience with air launching of the D-558-2, it is not expected that any particular stability problems will be encountered during this phase of the operation. Maximum normal acceleration will be required shortly after the drop in order to establish the desired flight trajectory angle. Adequate stability during the short period of subsonic flight immediately after launching will be assured by the large stabilizing tail surfaces which are required for high supersonic speeds, as discussed below.

6.1.2 Launching Control

As seen from the performance curves, typical flights cover about 500 miles, most of which is traveled in a space trajectory where it is impossible to change the direction of flight without large rocket devices. Since the airplane must land without power at a specific landing site, it is obvious that it must be aimed toward the landing base at launch. The need for control during powered flight is apparent for the following reasons:

1. The mother ship cannot aim the aircraft with sufficient accuracy.
2. Rocket thrust will not be sufficiently reproducible from flight to flight, either in magnitude or in alignment of thrust.

3. Changing winds and ambient conditions will introduce aiming errors.
4. Delays in starting the rocket motor.

The thrust is 50,000 lbs. or more. If the jet thrust were misaligned only 35 minutes a side force of 500 lbs. would be exerted. Thus the side force that may need to be counteracted could be rather large.

Several methods of changing the flight path during the powered period are:

1. By elevators and rudder
2. By the auxiliary coasting control jets
3. By controllable vanes immersed in the jet
4. By controlled separation in the rocket nozzle
5. By a gimbal-mounted rocket motor.

Elevators and Rudder

The dynamic pressure is of rather large magnitude during most of the powered flight, reaching a peak value of 870 lbs./sq.ft., but falling to about 25 lbs./sq.ft. at burnout. The mean dynamic pressure during this period is about 300 lbs./sq.ft. In the light of these figures it appears that adequate control can be provided by the elevators and rudder. Such a conclusion is consistent with the following facts.

- a. Elevators are used to pull the aircraft up from level flight into its trajectory. They are adequate.
- b. If early short flights indicate an appreciable thrust misalignment, the rocket motor can be rerigged on the ground, thereby minimizing the need for corrective type of control.
- c. The most precise control will be needed only on the all-out flights, flights which will not occur until the system is properly checked out and seasoned.
- d. Unlike a V-2, the aircraft is launched at a high forward speed, making the elevators and rudder more effective from the beginning.

Thus, even though the elevator control is too weak to overcome any bad misalignment, it need not be designed for such a severe

requirement because exploratory flights would have uncovered all design misalignments. Random deviations in thrust axis direction are small, being only about 2 minutes.

Auxiliary Coasting Control Jets

These jets are small, as will be described in Section 6.2, being designed only to produce slow changes in attitude outside the atmosphere. They produce a maximum side force of about 100 lbs. As this value is less than that provided by the rudder at burnout, and because they consume fuel when operated, they cannot be considered suitable.

Controllable Vanes in the Jet

The V-2 used this system. It provides powerful forces, but the heat and velocity are so great that to the best of our knowledge a reliable unit has never been made. Since it is unreliable, since it requires additional controls and weight, and since the conventional tail is shown to be adequate for air launching, such a method of control is rejected.

Controlled Separation in the Rocket Nozzle

Because vanes cannot be made to stand the rigors of the operation, some efforts have been made to provide control by using gas blasts through the side of the rocket nozzle to force a local separation within the nozzle and a consequent change of thrust direction. This method has received some study, notably by United Aircraft (Reference 11). The method has the great advantage of eliminating all obstructions within the jet. However, it requires holes through the walls of the nozzle, and possibly may introduce new cooling problems. The principle is new. It requires a control system to modulate the gas blasts, as well as an extra supply tank. Consequently at the present state of development the conventional elevator and rudder control appear superior.

Gimbal-Mounted Rocket Motor

Currently, the most satisfactory method of controlling the thrust direction is to mount the entire motor on gimbals and move it with respect to the airplane. There is some increase in weight, of course, but the system is reliable and subject to accurate design. The R.M.I. Model XIR30-RM-2 which is contemplated for this project, is designed for such a mounting. But because elevator and rudder control appears adequate, gimbal mounting appears unnecessary. If later work shows the need for more powerful control, then the gimbal mounted motor is the recommended choice.

6.1.3 Pullout Stability

Recovery from high altitude flight to a level attitude will be made at extreme Mach numbers, of the order of $M = 8.0$. This phase of the flight is expected to present the most serious stability and control problems.

Estimates of the static longitudinal stability and static directional stability have been made for a Mach number range of 1.2 to 6.0 using the best available theoretical methods, and the results are presented in Figures 15 and 16. The effects of the various components have been shown so that an appreciation of their contributions to the total stability may be obtained. The contributions due to the fuselage are large and remain relatively constant with Mach number. Since the lift curve slopes of the wing, horizontal and vertical tails decrease approximately as $1/\sqrt{M^2 - 1}$, their contributions to the longitudinal and directional stability decrease rapidly with Mach number. This results in the prediction of longitudinal and directional instability for the present configuration at a Mach number between 3.0 and 4.0.

Figure 17 shows the effect of increased horizontal tail size on the neutral point. It may be seen that the tail area must be increased considerably and, in fact, must approach the wing area in size if stabilization to the maximum flight Mach number is to be accomplished. Improvement in stability could also be obtained by moving the center of gravity forward from its present location at the leading edge of the wing to an aerodynamic chord. However, the gains from this source are limited by arrangement difficulties, space limitations, and by the danger of serious impairment of control during the landing phase.

There are several ways in which vertical tail area may be added to improve directional stability, as shown in Figure 18. However, the conventional method of increasing vertical tail area by placing additional fin area above the fuselage may introduce lateral-directional dynamic stability problems, due to the unfavorable inclination of the principal axis of inertia and the large aerodynamic rolling moment due to sideslip (dihedral effect). A preferred arrangement from the flying qualities viewpoint would consist of symmetrical upper and lower fins. The only way by which a full size lower fin could be employed, however, is to provide for folding until the parent aircraft is airborne and jettisoning prior to landing. A smaller ventral fin (as illustrated) could be employed by providing adjustable wing incidence so that the fuselage would be level during landing.

It is beyond the scope of the present proposal to investigate these problems to the extent necessary to determine an optimum solution

of the supersonic stabilization problem. Dynamic stability should be investigated carefully and the theory used in the present analysis should be verified by wind-tunnel tests before complete reliance is placed on the answers. Before a final configuration is established, the possibility of utilizing an automatic control system to overcome instability should be considered; if the tail areas are increased to the extent indicated for complete aerodynamic stability, severe weight penalties would be incurred.

6.1.4 Pullout Control

A detailed analysis has not been made of the control problems during the recovery phase, but because of the high Mach number and dynamic pressure, it is certain that power controls will be required for both lateral and longitudinal control. The lateral control would not be required to furnish extreme rates of roll since fighter-type maneuverability will not be a requirement. Therefore, design requirements as regards servo size and rate of motion will be moderate. For added safety, landing would be possible using manual control of both elevator and ailerons.

Since the longitudinal control must be capable of developing maximum airplane lift during both the recovery phase and after launching, the stabilizer must be the principle means of longitudinal control. However, fighter maneuverability requirements are not necessary, and a conventional slow-moving, adjustable stabilizer in combination with a direct control elevator system appears adequate. A fully-powered all-moving tail would provide more rapid control, and, in event wind-tunnel tests should show pitch-up tendencies near the stall, may be worthy of further consideration.

6.1.5 Landing

Stability and control characteristics during the landing phase are not expected to be critical. Based on past experience with comparable research type aircraft, a conventional control arrangement is believed to provide adequate controllability and maximum safety. In the subsonic speed range, control forces will be low enough to permit manual operation of controls and no reliance on power systems need be made.

6.2 Flight Outside the Atmosphere

During that portion of the flight which is outside the atmosphere, there are no external moments acting upon the airplane and thus there is no problem of instability. Control will be necessary, however, to correct for any residual moments induced during the powered ascent, to permit a clean attitude of re-entry into the atmosphere, and to allow the pilot freedom of orientation during

the space trajectory. The amount of control needed for these maneuvers is indefinite, but apparently it will be small. Several methods for providing control in space are discussed below.

Control by Flywheels

From a control systems standpoint flywheel control is the most desirable, but it appears to be too heavy for the present application. The method utilizes the principle of conservation of angular momentum. For a given axis of rotation, let

I_A = Moment of inertia of the airplane

I_F = Moment of inertia of the flywheel

ω_A = Angular velocity of the airplane

ω_F = Angular velocity of the flywheel

Then since the airplane is coasting in a vacuum where no outside forces can be applied,

$$\omega_A I_A + \omega_F I_F = 0$$

That is
$$\omega_A = - \frac{I_F}{I_A} \omega_F$$

Hence, when an angular velocity ω_F is imparted to the flywheel the airplane will rotate in the opposite direction with a velocity proportional to the ratio of the moments of inertia. Furthermore, since

$$d\theta = \omega dt$$

then
$$d\theta_A = - \frac{I_F}{I_A} d\theta_F$$

Therefore the total change in angle of the airplane is a fixed multiple of the angular change of the flywheel. For instance if $I_F/I_A = 1/1000$ then 1° pitch change would require a 1000° rotation of the flywheel, or about 3 revolutions. In this system three flywheels rotating about the three axes will be needed if control about all three axes is sought.

Control by Gyroscopes

A gyroscope on frictionless supports will tend to maintain its orientation in space. Therefore, it can serve as a support against which to push in order to turn the airplane. However, flywheel strength considerations prevent it from absorbing or transmitting any more momentum than the flywheels of the previous method. Furthermore, precession would seriously complicate the control system and the gimbal mount would be heavy.

Control by Jets

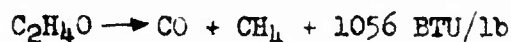
A system of jets would consist of six jets to produce control about all three axes. Using a conservative value for specific impulse of 100, the system proves to be far lighter than the flywheel method, unless it is to be used over long periods of time. The method to be discussed is mechanically rather simple; its greatest drawback is that it controls angular acceleration instead of angular velocity. Therefore, because position is given by the double integration of acceleration as against the single integration of velocity, accurate position control using jets is more difficult.

A system combining simplicity and control sensitivity is as shown in Figure 19.

The system illustrated will handle a flow rate of about 1 pound per second of H_2O_2 , providing a maximum thrust of about 100 lbs. A 25 pound supply has been assumed arbitrarily. The H_2O_2 supply is kept in a pressurized tank at about 500 psi, and a feed regulator valve allows it to flow into the catalyst chamber whenever the back pressure falls below 400 psi. Hence a supply of steam at 400 psi will always be available, which is piped to the six nozzles. Whenever control is needed the appropriate needle valve is operated. If the needle is so shaped that the throat area is proportional to the control stick movement, proportional control is provided and any amount of thrust from zero to 100 lbs. will be available.

The system sketched in the figure has approximately the correct proportions, assuming H_2O_2 at 400 psi is used. It is quite small and compact except for the large 1 inch line from the reaction chamber to the nozzles. These lines will contain hot steam, and if the heat loss is appreciable, excess H_2O_2 will be required to make up for the condensate. Because the gas generated is steam and oxygen, little pressure would exist if the lines become cold, for all the steam would condense to water.

With ethylene oxide, this problem would not be nearly so severe. Its decomposition reaction is



Every mole of $\text{C}_2\text{H}_4\text{O}$ (44 grams) generates 4480 cubic centimeters of gas at one atm. pressure. These products remain as gases even at low temperatures because they both have very low boiling points. The boiling points at atmospheric pressure for the decomposition products are

H_2O	B.P.	100° C.
O_2		-183° C.
CO		-190° C.
CH_4		-161.5° C.

This feature, plus the fact that $\text{C}_2\text{H}_4\text{O}$ inherently possesses a higher theoretical specific impulse, makes it the preferred choice provided a suitable light gas generator can be produced. The reason that an H_2O_2 system is shown in the diagram is that it is a more tried and proven system, and provides conservative weight estimates.

Figure 20 throws light on the comparative weights of the flywheel and jet methods of attitude control. The flywheel was assumed to be a steel torus, rotating at a peripheral speed of 1000 ft/sec. The abscissa represents the angular rotation that must be imparted to the airplane. For example if a change in rate of rotation of 0.1 rad. per sec. is sought a 19" diameter flywheel will be required. In order to produce the change it must change its rotational speed by 12,000 RPM. The wheel alone will weigh about 123 lbs. To do the same job with jets acting at 12 ft. from the center of gravity, only 1.7 lbs. of propellants will be needed. The jet system, of course, consumes mass every time it produces thrust. But if the controls are used with moderation, the jet system appears to be much the lighter.

A final advantage of the jet system is that it can be integrated with the auxiliary power unit, for both require the same mono-propellant at roughly the same pressure. This feature will permit use of a common reaction chamber, etc., that should save weight and simplify the design problems.

6.3 Attitude Sensing at Re-entry

In re-entering the atmosphere the aircraft should have the proper angle of attack and be essentially unyawed to avoid excessive abnormal loads. This will require a pitch and yaw indicator to provide the pilot with the maximum time to point his craft into the relative wind; for even though the pullout may be automatically programmed it is assumed that the pilot is responsible for making a clean entry into the atmosphere.

Hence we have the problem of designing a pitch and yaw indicator capable of sensing exceedingly low forces or pressures, but capable of withstanding the maximum dynamic pressures encountered during the complete pullout. The instrument need not be precise, for it is only to serve as a guide for pointing the nose into the wind at heights where a pilot may otherwise lose all sense of orientation. Thus the demands differ considerably from those for a pitch or yaw indicator as commonly used in flight testing. Several possible methods are suggested.

- (a) A weathervane - either direct or remote reading.
- (b) A pitch or yaw indicator which measures the relative Mach number or pressure ratio on opposite sides of a symmetrical sphere, cone or other convenient shape.
- (c) A vane inside a conventional instrument case that indicates the direction of the resultant momentum from two jets of air brought in from a pair of symmetrical external tubes.
- (d) A gauge similar to the Reichardt gauge, to be described.

The first two do not seem very satisfactory. Any external weather-vane will undergo very high loads, necessitating such rugged construction that it will be too insensitive at high altitudes where it is needed. Direct reading is not too desirable because the pilot would have to watch two areas at once - his cockpit instrument panel and the external vane.

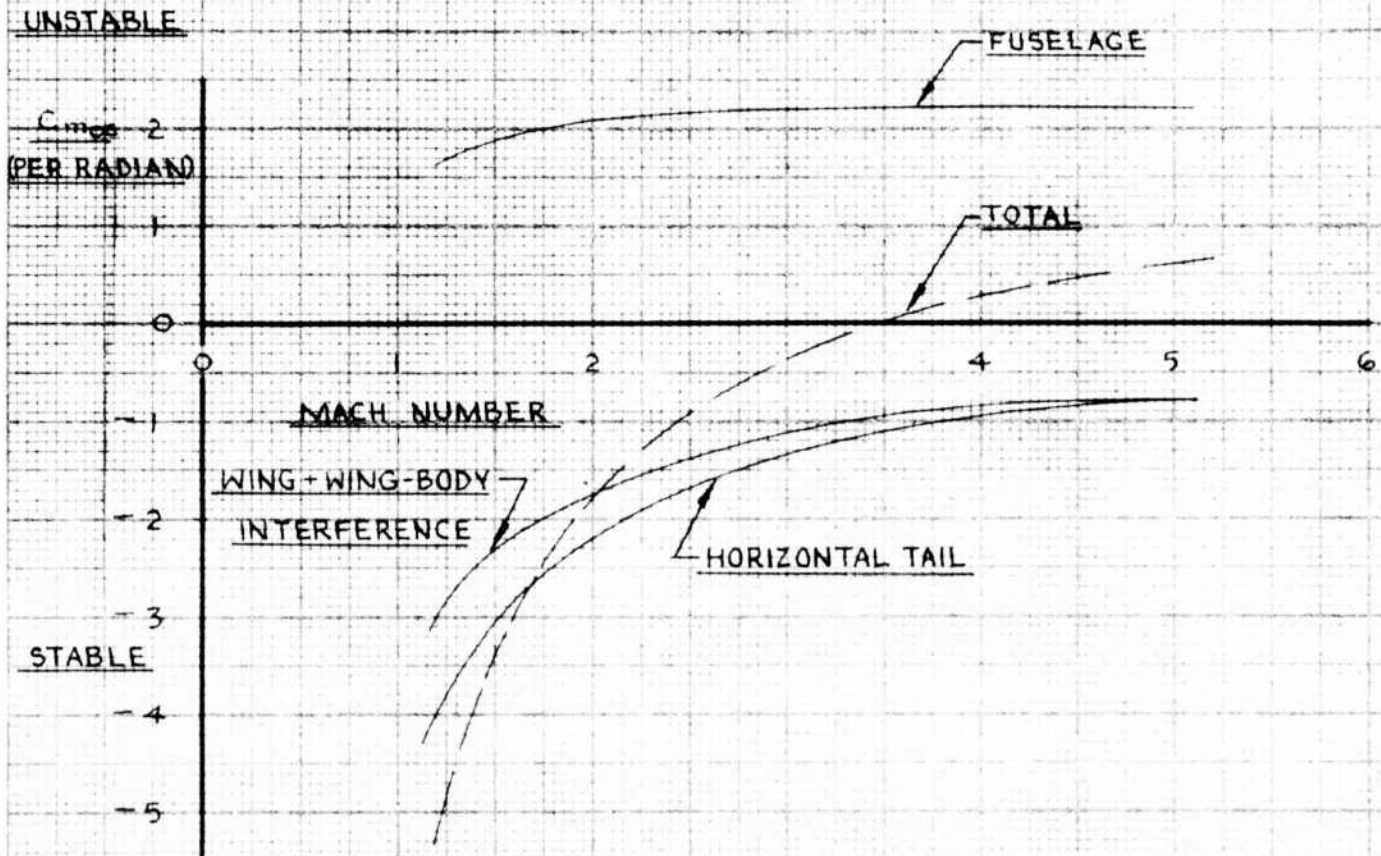
A sphere, cone, or any other symmetrical shape will produce different pressures on the opposite sides unless it is pointing directly into the wind. This feature can be used as the basis for a pitch or yaw indicating system. Pressure differentials cannot be used, because 1° yaw at 1000 knots indicated airspeed represents a much greater differential than 1° yaw at 1 knot; thus the system must utilize a pressure ratio. However it is doubtful that this method would be sensitive enough for the purpose at hand and rugged enough to withstand the maximum pressures.

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MODEL 671

CONTRIBUTION OF VARIOUS COMPONENTS TO THE LONGITUDINAL STABILITY

CG AT ZERO PERCENT M.A.C.

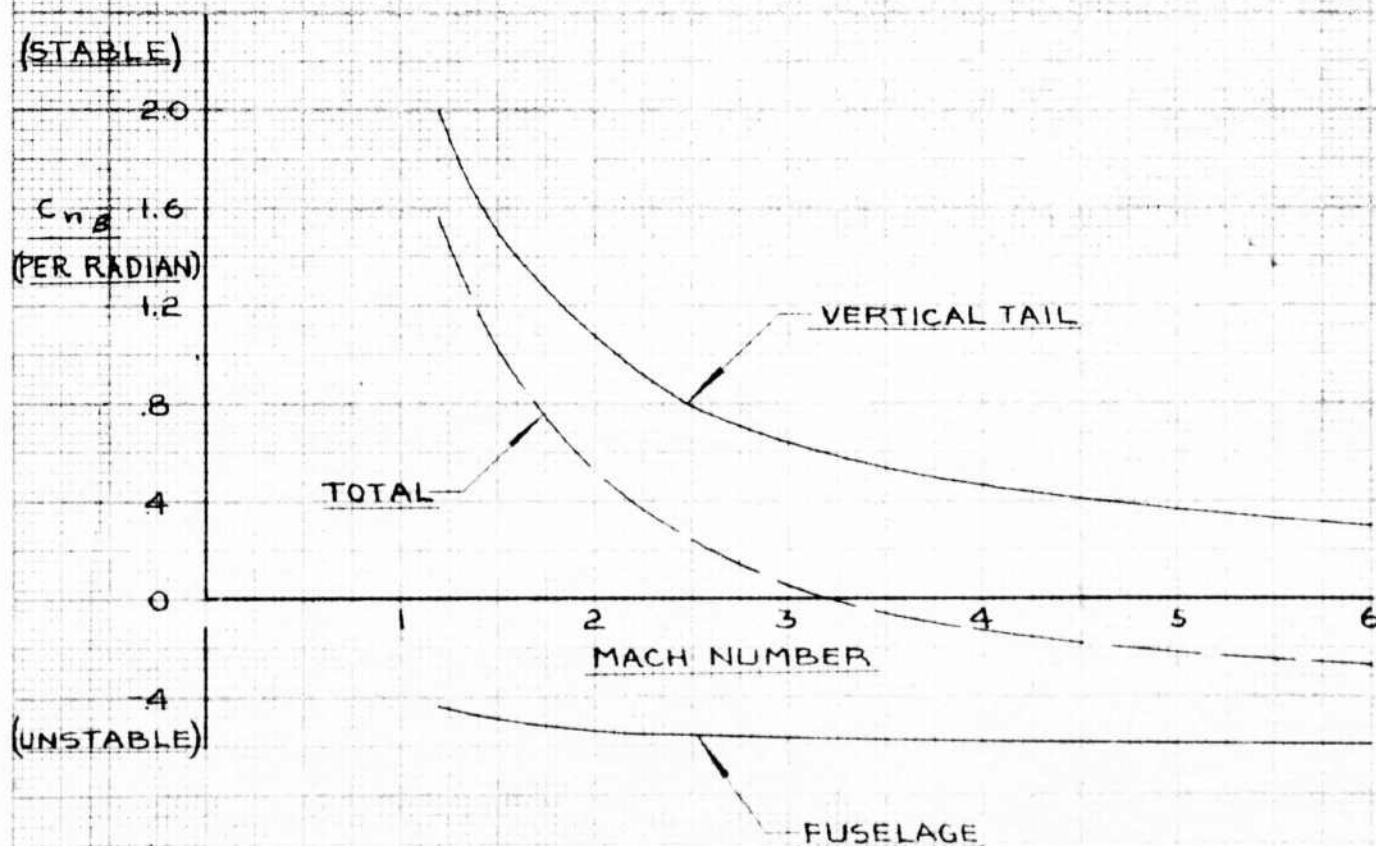


DOUGLAS AIRCRAFT COMPANY, INC. EL SEGUNDO PLANT EL SEGUNDO, CALIFORNIA

MODEL 671

CONTRIBUTION OF VARIOUS COMPONENTS TO THE DIRECTIONAL STABILITY

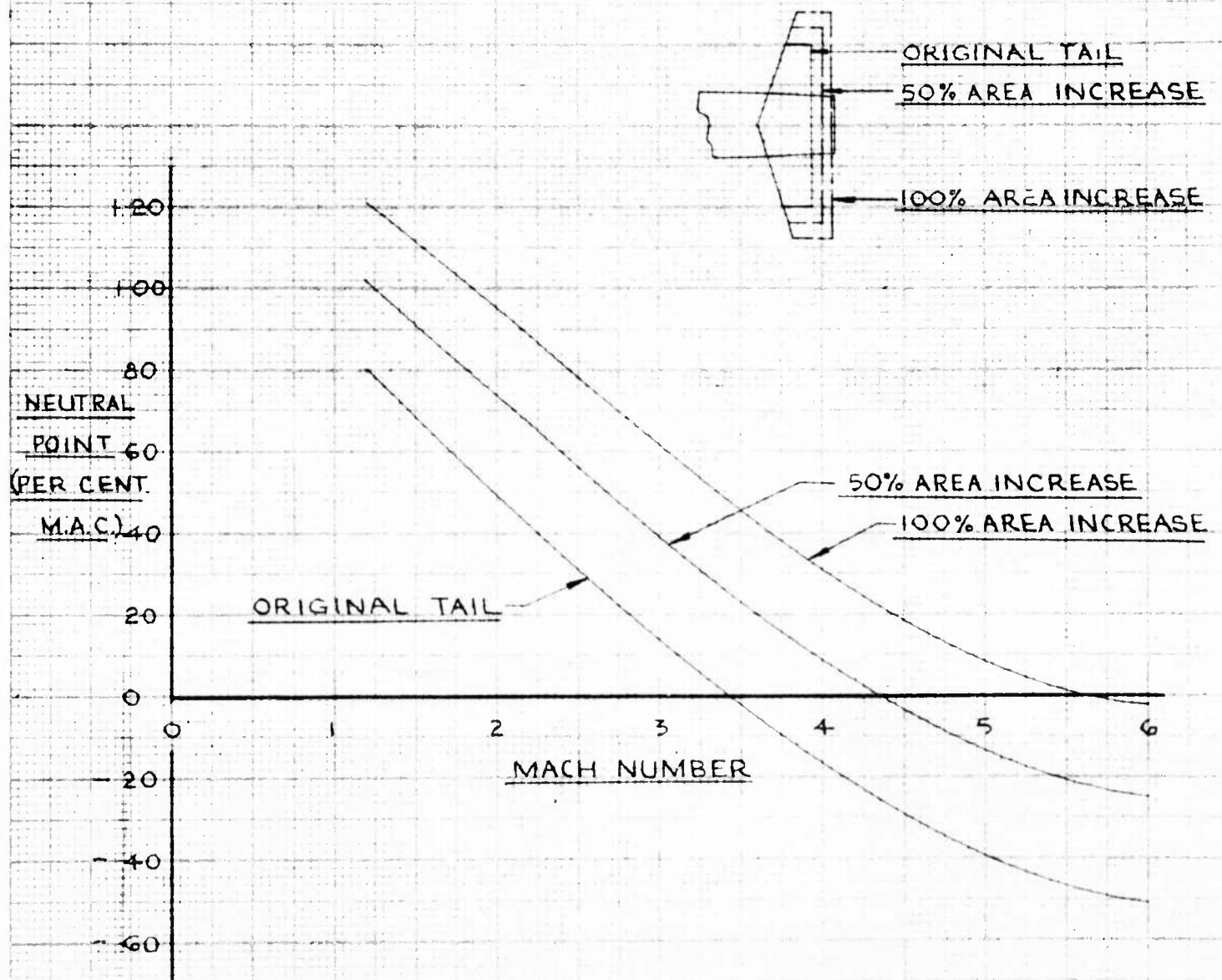
C.G. AT ZERO PER CENT M.A.C.



DOUGLAS AIRCRAFT COMPANY, INC. EL SEGUNDO PLANT EL SEGUNDO, CALIFORNIA

MODEL 671

EFFECT OF HORIZONTAL TAIL SIZE ON LONGITUDINAL STABILITY

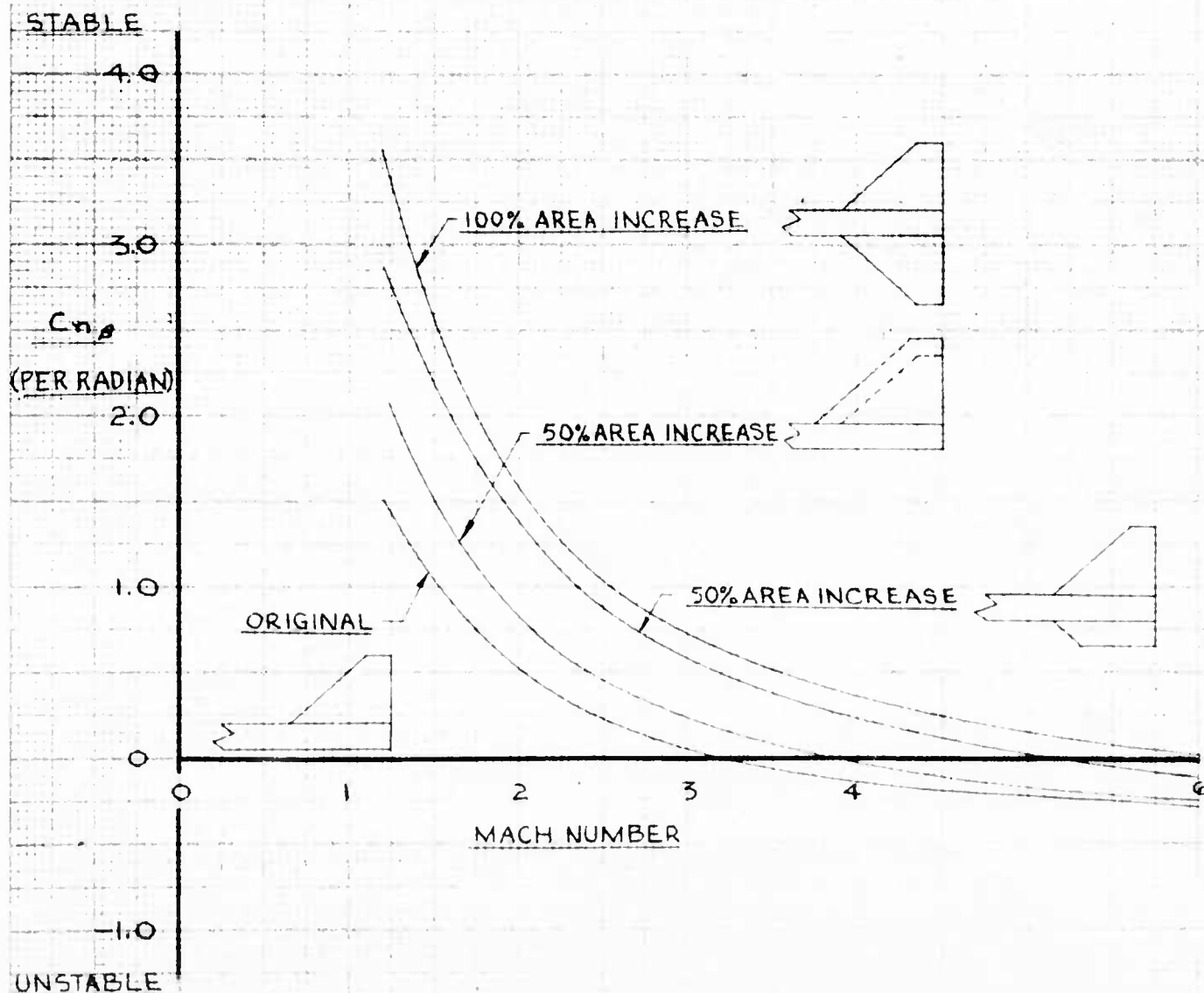


DOUGLAS AIRCRAFT COMPANY INC. EL SEGUNDO PLANT EL SEGUNDO CALIFORNIA

MODEL 671

**EFFECT OF VERTICAL TAIL SIZE ON
 DIRECTIONAL STABILITY**

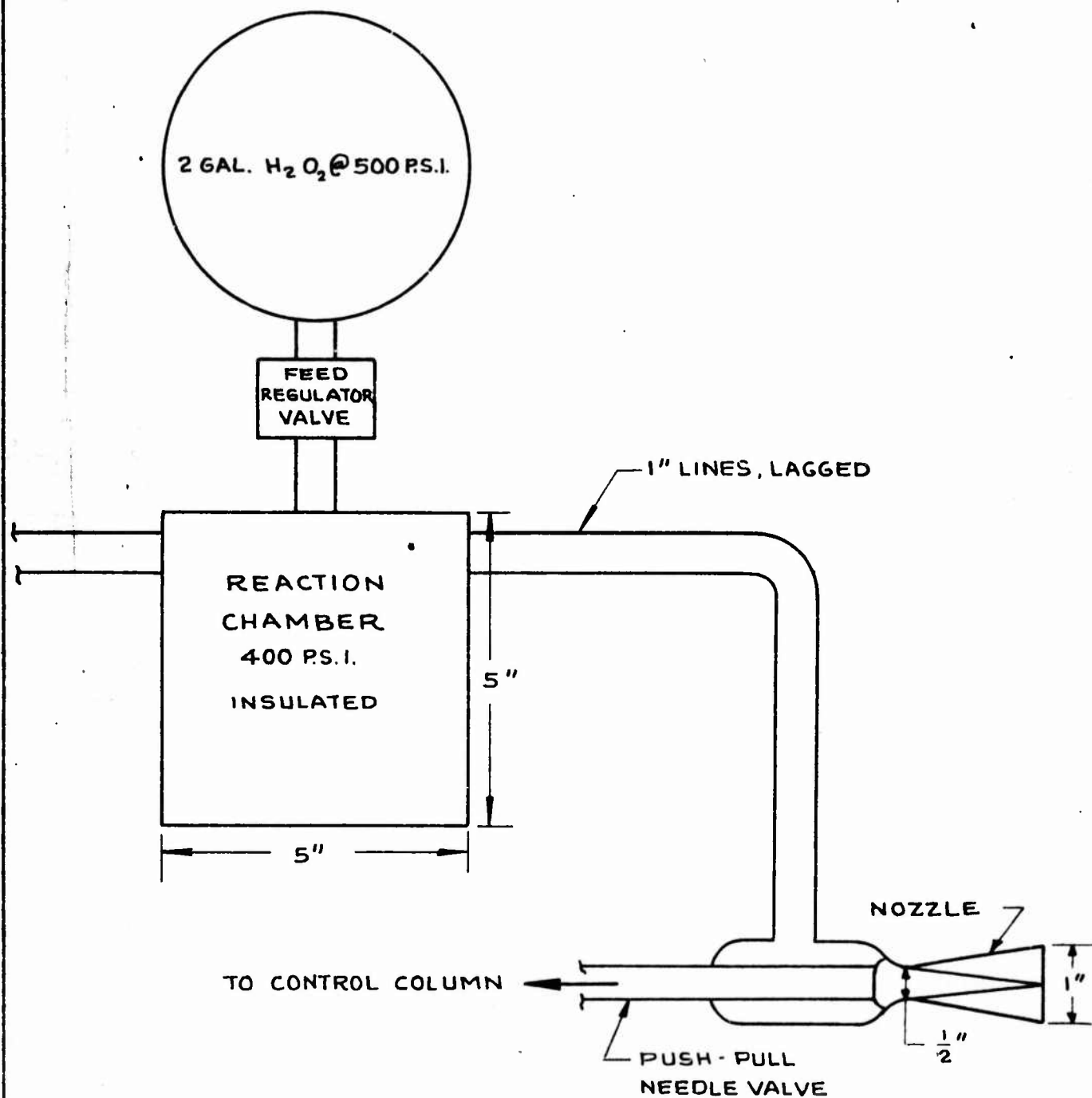
C.G. AT ZERO PERCENT M.A.C.



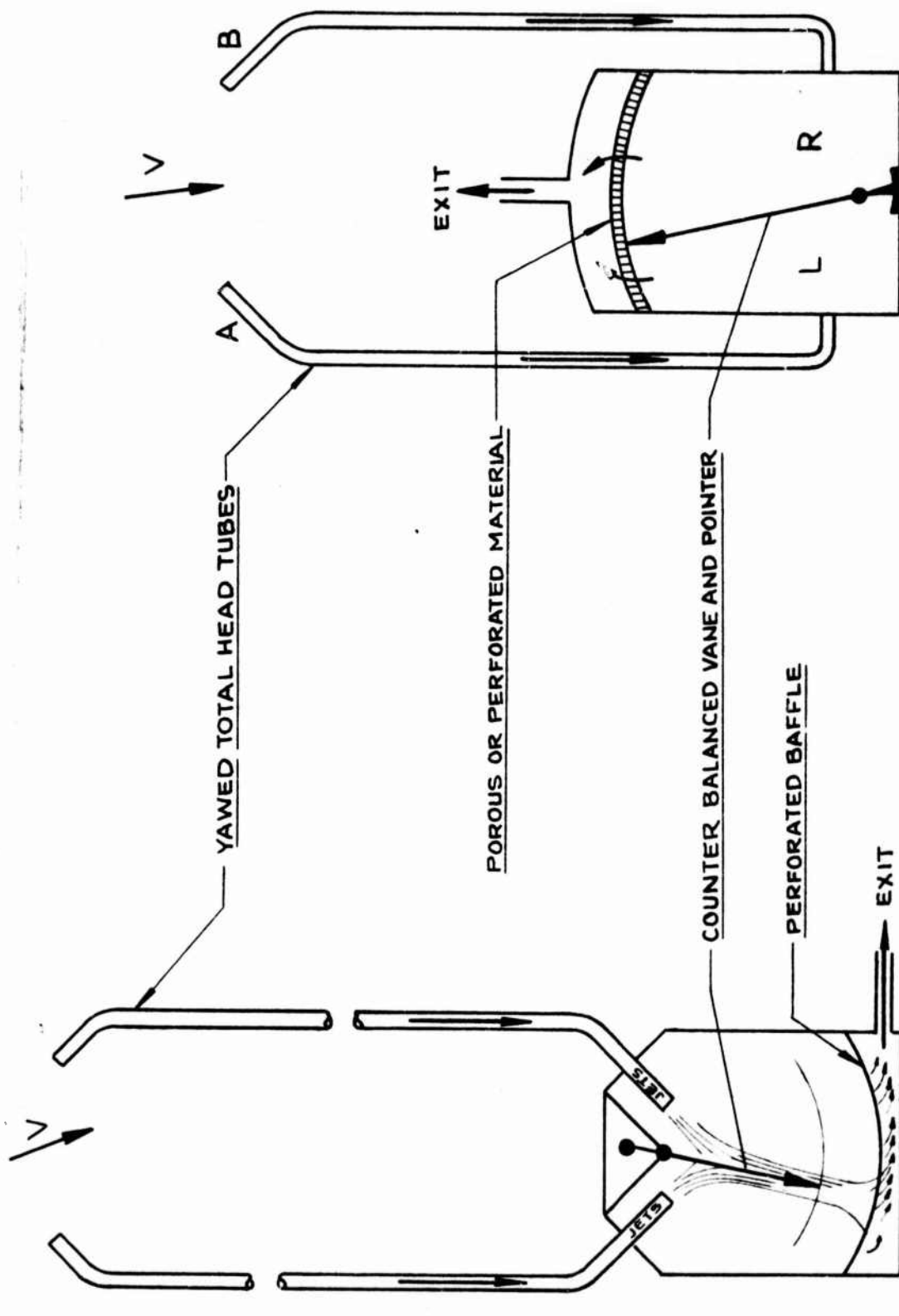
DOUGLAS AIRCRAFT COMPANY, INC. EL SEGUNDO DIVISION EL SEGUNDO, CALIFORNIA

MODEL 671

AUXILIARY JET CONTROL SYSTEM



MODEL 671 ATTITUDE SENSING DEVICES



(a) MOMENTUM TYPE

(b) PRESSURE RATIO TYPE

7.0 HEATING PROBLEM

7.1 Basic Heat Transfer Methods

It is impractical in the present study to make a complete survey of the temperatures expected on the airplane. For the type of flight which is planned, the calculations are quite complicated and tedious to obtain reasonable estimates of the skin temperatures; in addition, the application of the basic theories is very questionable for the range of variables involved. These difficulties will be discussed in detail below.

Therefore, detailed temperature calculations have been restricted to the evaluation of the transient skin temperature at two typical stations on the airplane. From these results the temperatures at other locations can be estimated roughly, and an overall picture of the heating problem obtained for general design purposes. The specific conditions chosen were as follows:

- (a) A point one foot downstream of the leading edge of a flat plate, with a turbulent boundary layer.
- (b) At a two-dimensional stagnation point.

In both cases it has been assumed that the skin is 0.060 inch stainless steel, perfectly insulated on the inside, with no conduction in the plane of the surface.

The temperatures were calculated by the standard step-by-step method which is common to problems involving highly transient conditions. Equating the net heat flux into a unit area of the surface to the heat absorbed by the surface, one gets

$$w_s c_s y_s \frac{dT_s}{d\theta} = h (T_{aw} - T_s) + \alpha G_s - \epsilon \sigma \left(\frac{T_s}{100} \right)^4$$

The left-hand term represents the rate of heat absorption by the surface. The first term on the right is the rate of heat transfer by convection from the boundary layer, the second term represents solar heating, and the last term is the heat lost by radiation from the surface.

If the differential equation is replaced by a finite difference equation, the change in surface temperature in an incremental time, $\Delta\theta$, is given by:

$$\Delta T_s = \frac{\Delta\theta}{w_s c_s y_s} \left[h (T_{aw} - T_s) + \alpha G_s - \epsilon \sigma \left(\frac{T_s}{100} \right)^4 \right]$$

For the calculations, stainless steel has been assumed as the skin material, and it is not believed that the use of other comparable metals will change the results significantly. Physical and thermal properties have been obtained from standard handbooks. The emissivity (ϵ) and solar absorptivity (α) of the surface have been chosen arbitrarily as 0.90 and 0.15, respectively.

The heat flux from the sun (G_s) is essentially constant above 50,000 ft.; the value given in Reference 12 has been used.

7.2 Skin Temperatures

The heat transfer coefficient and adiabatic wall temperature at the one foot station have been calculated on the basis of turbulent boundary layer flow with Mach number and wall temperature corrections as given by the theory of Van Driest. The choice of a turbulent boundary layer and the effect of Mach number on heat transfer represent the most critical factors in the determination of the transient skin temperature. Unfortunately, it is extremely difficult at present to make reliable predictions of transition from laminar to turbulent flow, or of the effects of compressibility on the boundary layer, even at low supersonic speeds. Extension of these doubtful theories to the very high Mach numbers realized in the present flight path represents pure hypothesis, and verification or modification of the results must await future tests. The general status of current boundary layer theory is discussed briefly in a later paragraph. The heat transfer coefficient at the stagnation point was calculated by standard formulas (Reference 13). The increase in total temperature in passing through the normal shockwave at the nose was included.

Temperatures versus time, calculated with these assumptions, are shown in Figures 22 and 23 for the 0.060 inch stainless steel skin for several different ballistic flight paths.

These results can be used to provide rough estimates of the temperatures at other stations on the airplane, except at times of rapidly changing temperature. For conditions which are fairly well stabilized, the difference between the surface temperature and the stagnation-point temperature will vary approximately as the one-fifth power of the distance from the stagnation point, other variables being constant (Figure 24).

It should be repeated that these results are calculated with the assumptions of no heat transfer to the interior and no heat conduction in the plane of the surface. Aside from the basic uncertainties regarding the heat transfer coefficient, etc., the above assumptions tend to make the results conservative. Thus

heat transferred to large interior heat sinks such as bulkheads or cooled compartments will lower the local skin temperatures, often by very appreciable amounts. Conduction in the plane of the skin will have a tendency to equalize the temperature over the surface, which can be especially important in reducing temperatures in the vicinity of the leading edge or in regions of large surface temperature gradients. These refinements could be included in a more detailed temperature analysis, but, because of the added complexity, would probably be attempted only in regions which created critical problems in design, such as the leading edge in the present case.

It is apparent from the magnitude of the temperatures shown in Figure 22 that some means of lowering the peak temperature by, say, 500° F or more over the major part of the surface is a necessity in order to obtain a practical structural weight for the airplane. To establish the effect of applying insulation to the outside of the skin, transient skin temperatures were calculated using several thicknesses of a "good insulating material" (i.e., one which has a low thermal diffusivity). The assumed properties of the insulation were: $k = 0.05 \text{ BTU/hr ft}^2\text{F}$, $c_p = 0.20 \text{ BTU/lb}^{\circ}\text{F}$, $w = 20 \text{ lb/ft}^3$. This calculation is extremely tedious, even with the gross assumption of a linear temperature gradient through the insulation, since it is necessary to choose time intervals of the order of one-quarter of a second to prevent divergence of the finite difference solution. The results of several such calculations are shown in Figure 25, where it will be observed that the use of a small thickness of insulating material does indeed reduce the peak temperature by a very satisfactory increment. The practical problems connected with the development and application of such a material may be formidable, but it is apparent that the reward is appreciable.

Other possibilities for reducing the surface temperature do exist, and remain to be investigated in detail. The injection of a cool gas, such as bottled oxygen, into the boundary layer has been investigated by others as a possible means of reducing convective heat transfer, with mixed results. Recent theoretical investigations (Reference 14) have indicated that the injection into the boundary layer of a gas with substantially different thermal properties can have a large effect on the heat transfer, over and above the purely physical effect of a cool injection. The mechanical problems connected with any type of boundary layer injection are not minor, however, and a thorough study is necessary to establish the net worth of the various techniques.

7.3 Interior Temperatures

Interior insulation need be considered only for those compartments in which the temperature must be maintained at particularly low

values. The resistance to heat flow afforded by the stagnant air (or by the vacuum) inside the airplane will be sufficient in most cases to cause an appreciable time lag and thereby reduce the peak interior temperatures by large amounts. However, particular attention must be given to the thermal environment of the pilot, the instrumentation, and the fuel. The latter can be dismissed forthwith as a major problem, since all usable fuel has been expended by the time any critical temperatures are realized. A minor problem is presented by the residue of fuel which inevitably remains after burnout. If the fuel tanks cannot be completely evacuated in some manner, then the remaining fuel must be prevented from reaching its flash point temperature, or it must be established that insufficient oxygen will be present to support spontaneous combustion.

Special attention must be given to instrumentation temperatures for two reasons. Most standard instruments are reliable through a restricted ambient temperature range only, and nearly all electrical connections both inside and outside of the instruments are made by soldering. Standard soft solder has a melting point of about 360°F , so that it may be necessary to specify the use of silver solder (melting point 1300°F) in all equipment. However, since it is doubtful if the majority of standard instruments are reliable above 360°F , it will probably be required to either build special instrumentation, or insulate the equipment sufficiently to keep the maximum ambient temperature quite low. Of course, the usual methods of cooling electronic compartments by ducting ram air are useless in the present application, where the ram air temperatures are of the order of several thousand degrees.

7.4 General Considerations

The net heat flow into any surface is composed of the heat transferred by convection from the boundary layer, the irradiation to the surface from the sun, and the radiant heat lost by the surface. The relative importance of these heating effects changes drastically, depending on the flight regime. Thus, for flight at nominal speeds and altitudes (say, below 100,000 ft. and $M = 2$) the heat loss due to surface radiation effectively cancels the gain due to solar heating, and, since both are small relative to the convective heating, these contributions can be neglected with safety. At extremely high altitudes, the absence of air eliminates the convective heating, and the problem becomes one of radiation only. At very high speeds within the atmosphere, the heating due to convection is dominant initially, but as the surface heats up, surface radiation rapidly becomes of the same order of magnitude. In the present flight plan, all three flight regimes are encountered, and it is simplest to consider all three contributions

throughout the calculations.

The only feature of radiant heat transfer over which the designer has any control is the emissivity and absorptivity of the surface. In the present case, it is desirable to reduce the solar absorptivity of the surface as much as possible, in order to lower the equilibrium temperature which is approached during the long period of flight outside the atmosphere. It is also required that the emissivity of the surface be as high as possible, both to lower the vacuum equilibrium temperature and to increase the lag in temperature rise during the final re-entry. In general, these demands appear to be contradictory, since emissivity and absorptivity are equal for the same radiation-source temperature. However, solar irradiation comes from a source at an effective temperature of about $10,000^{\circ}\text{F}$, whereas the surface radiation has a source temperature of about 2000°F or less. Some degree of independence is possible, then; a study of published values for emissivity indicates that a material such as white paint, say, has emissivities of around 0.9 in the lower temperature range and about 0.15 for effective solar temperatures. It remains to establish similar values for a material capable of withstanding temperatures of the order of 2000°F , but at the present time this would appear to be a relatively minor developmental problem.

7.5 Boundary Layer Theory

It is unfortunate that the largest contributing factor to the high temperatures of re-entry, the convective heating from the boundary layer, is the one about which there is the least knowledge. To determine reliably the heat transfer from the boundary layer, it is necessary to know first the type of boundary layer flow, whether laminar, turbulent, or transitional, and then to know the characteristics of that type of flow.

Transition from laminar to turbulent flow is known to depend on the Reynolds number, Mach number, surface temperature, pressure gradient, and surface condition, but the quantitative effects of each of these parameters has not yet been established even for moderate flow conditions. The basic mechanism of transition is not fully understood, and it has proved to be difficult to isolate the effects experimentally. In general, theory and experiment indicate that transition occurs at a local Reynolds number of 2.8×10^6 on an unheated flat plate at low speed, and that it moves downstream with decreasing free stream Reynolds number, increasing Mach number, decreasing surface temperature, a favorable pressure gradient (i.e., decreasing pressures in the direction of flow), and decreasing surface roughness. Obviously, it is practically impossible to make a reliable prediction of transition at the

present time for the Mach numbers expected on re-entry into the atmosphere.

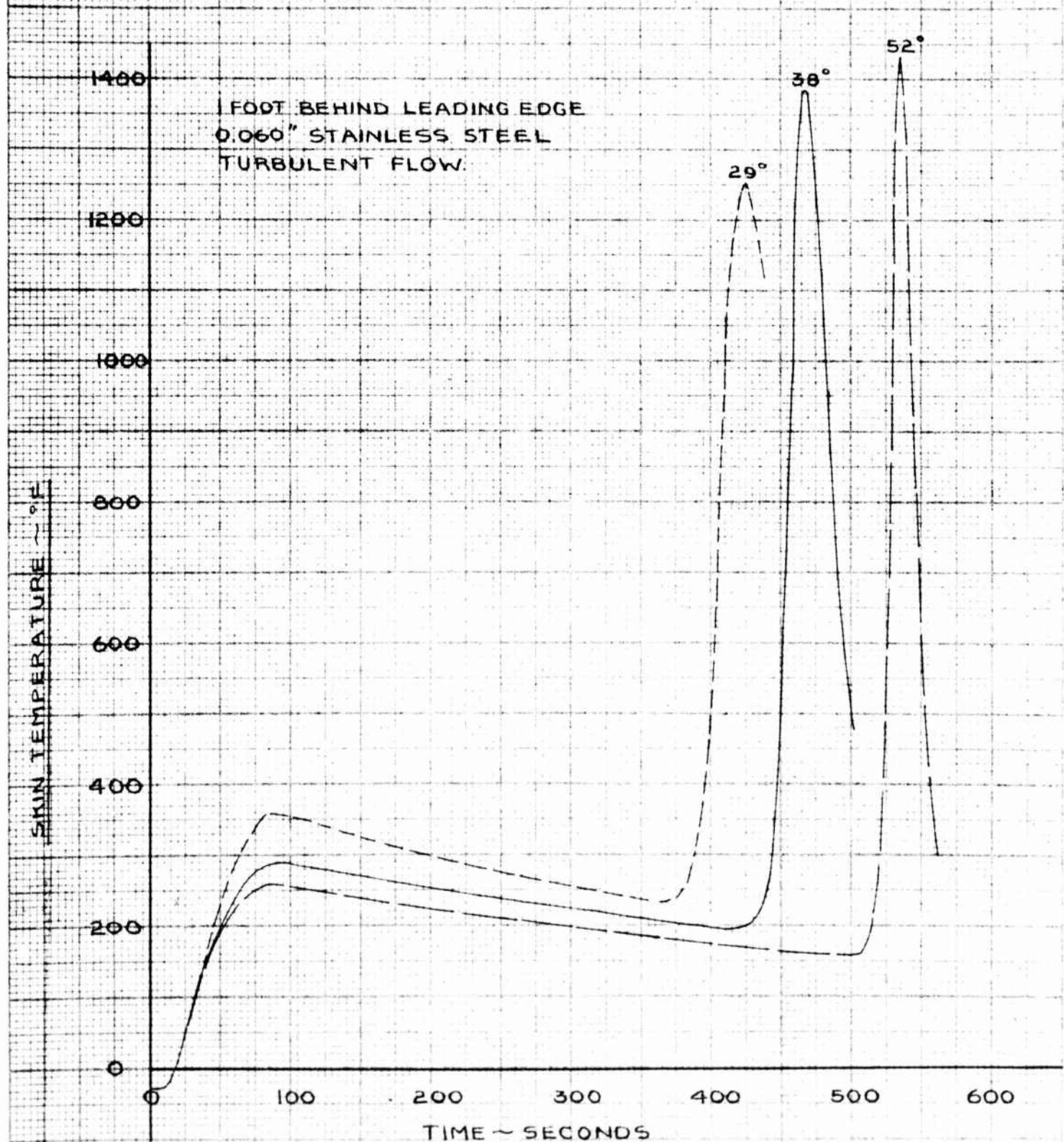
Heat transfer from a laminar boundary layer has been well established theoretically in recent years. Crocco (Reference 7), and later Van Driest (Reference 8), have solved exactly the boundary layer equations for compressible flow over a flat plate with heat transfer, and their results have checked well with experiment. The completely general problem of compressible laminar flow over an arbitrary body with heat transfer has been solved in an approximate manner (Reference 15), which makes it possible to predict the simultaneous effects of specific pressure gradients and surface temperature gradients.

The corresponding turbulent heat transfer problem is in a considerable state of confusion. Since the basic mechanism of turbulence has not been established, all theories are semi-empirical in nature. The low speed flat plate case has been developed to the point where good agreement with experiment can be expected. Extension to compressible flow is not straightforward, and a number of widely divergent methods have been suggested, many of which agree reasonably well with the limited amount of experimental data available at moderate supersonic speeds. It has been shown, however, that extrapolation of these theories to Mach numbers of the order of those of the present flight paths can lead to differences between various theories of several hundred percent (Reference 9). The possible effects of pressure gradient and surface temperature gradient are completely unpredictable under these conditions. For the present calculations, the theory of Van Driest for a flat plate (Reference 10) has been extrapolated for use in the present Mach number range, not because of any basic superiority of this theory over the others, but because it agrees with the data at lower Mach numbers and the theoretical results are readily available. The use of another theory, say that of Li and Magamatsu (Reference 16), would yield heat transfer coefficients at least twice as large at a Mach number of 8.

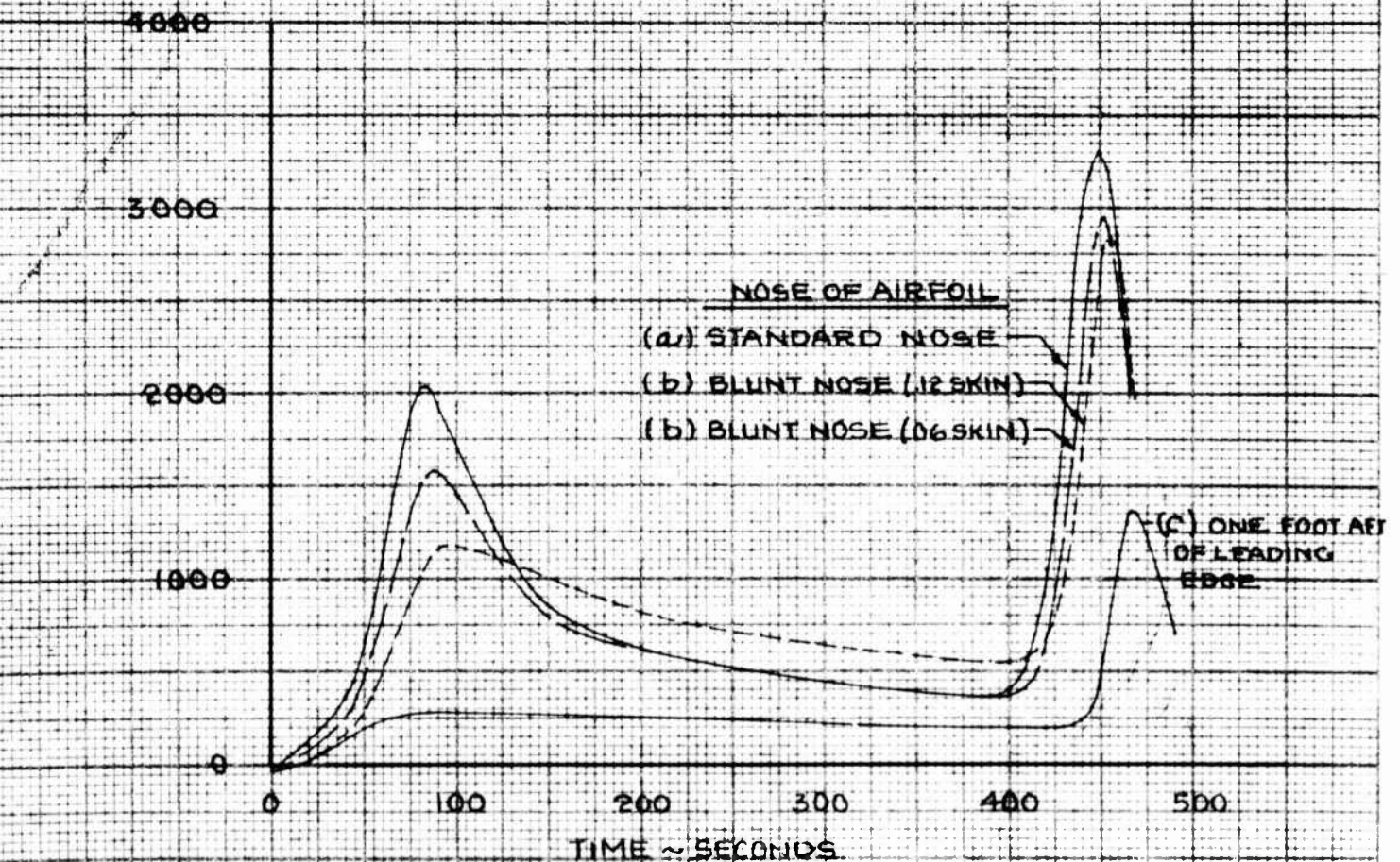
DOUGLAS AIRCRAFT COMPANY, INC. EL SEGUNDO PLANT EL SEGUNDO, CALIFORNIA

MODEL 671

TRANSIENT SKIN TEMPERATURE FOR VARIOUS FLIGHT PATHS



DOUGLAS AIRCRAFT COMPANY, INC. EL SEGUNDO PLANT EL SEGUNDO, CALIFORNIA

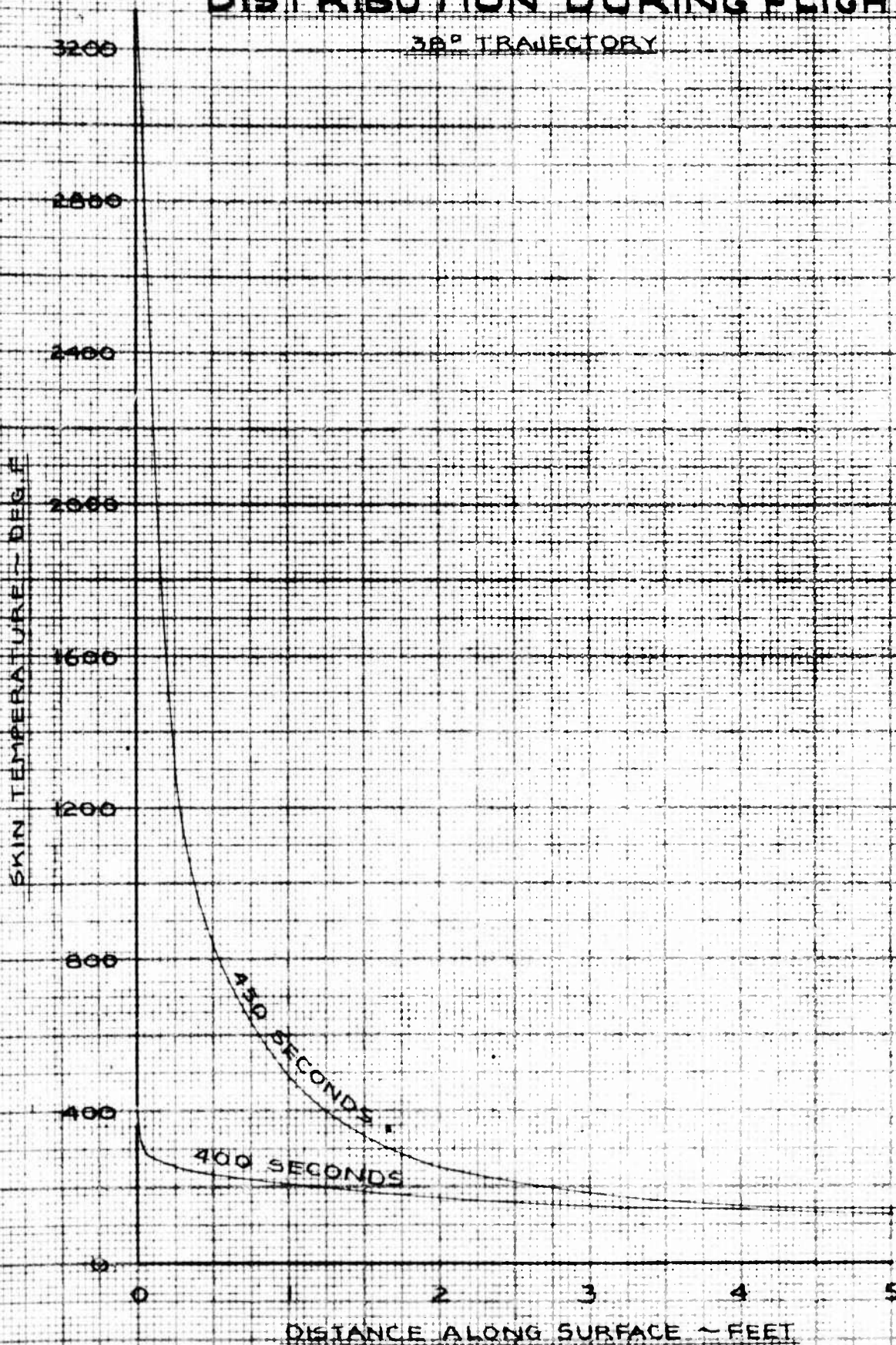
MODEL 671SKIN TEMPERATURES AT LEADING EDGETURBULENT BOUNDARY LAYERTHERMAL PROPERTIES FOR 0.060 STAINLESS STEEL65A006 AIRFOILSTANDARD NOSE RADIUS = 0.12 IN.BLUNT NOSE RADIUS = 0.50 IN.SKIN TEMPERATURE $^{\circ}\text{F}$ 

DOUGLAS AIRCRAFT COMPANY, INC. EL SEGUNDO PLANT EL SEGUNDO, CALIFORNIA

MODEL 671

**ESTIMATE OF TEMPERATURE
DISTRIBUTION DURING FLIGHT**

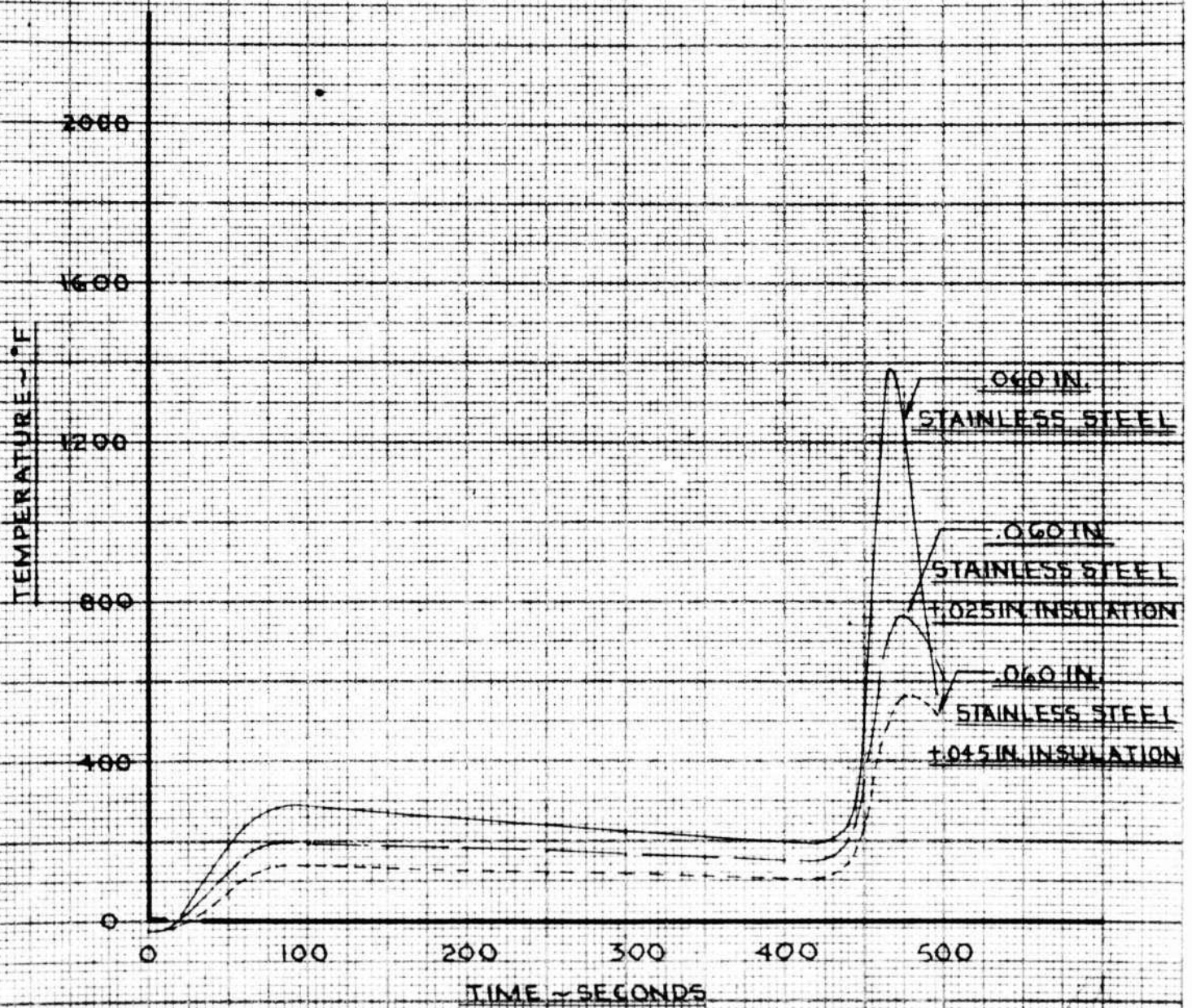
30° TRAJECTORY



DOUGLAS AIRCRAFT COMPANY INC EL SEGUNDO PLANT EL SEGUNDO CALIFORNIA

MODEL 671

**EFFECT OF EXTERIOR INSULATION ON
SKIN TEMPERATURES
ONE FOOT AFT OF LEADING EDGE
TURBULENT BOUNDARY LAYER**



8.0 PROPULSION

8.1 Power Plant

The requirements for the power plant are well defined within rather narrow limits by the minimum acceptable airplane performance on the one hand, and the limitations imposed by the severe pullout restrictions on the other. Therefore, it was determined early in the study that a desirable rocket propulsion unit should produce about 50,000 pounds of thrust with fuel consumption of about 200 pounds per second, and be as light in weight as possible. In addition, since new rocket power plants require long developmental periods, any rocket engine which is considered should be available within a year or two so that the project can remain on a practical basis.

These requirements simplify the propulsion problem, since all of the present production rocket power plants are too small, and most of those rated to produce sufficient thrust are in the form of long-term research projects.

Fortunately, one unit under development almost exactly meets the specifications: the Reaction Motors Model XLR30-RM-2 unit, producing 50,000 pounds of thrust at sea level (Reference 17). The unit is light, uses liquid oxygen and ammonia, and develops a sea level specific impulse of 245 seconds at full thrust. Furthermore, the design is such that it can easily be modified to burn the more powerful hydrazine if future developments justify the change. As a result, all design studies immediately centered around this unit as the power plant. Its tentative specifications are given in Table III.

The nozzle geometry has been examined in order to correct the thrust for the effect of altitude. The correction is appreciable, as shown in Figure 26; the thrust increases to about 59,000 lbs. at high altitudes.

As discussed previously, the re-entry problem is critical for this aircraft. The above power plant is easily capable of propelling the airplane to altitudes above those for safe recovery. Therefore there is little need to search further for an improved power plant for the present project, since profitable use of better power plants can not be made until the performance is raised to the satellite level.

8.2 Propellants

A number of reports by Rand, the NACA, and the Caltech Jet Propulsion Laboratory were consulted in order to gain a general

picture of the propellant field. Aside from handling and storage problems, two of the most important properties of a propellant are high energy, i.e. high values of the specific impulse, and high density. High density is of direct value because it allows reduction in tank sizes, pump sizes and weights. Figure 27 is a useful plot of these two properties at sea level for a variety of propellants. On the basis of density, hydrogen is seen to be a very poor fuel. Hydrazine (N_2H_4) is one of the better fuels because of its high density. Of the oxidizers only two have a sufficient background of development to be considered for the present project: liquid oxygen and nitric acid - either red fuming (RFNA) or white fuming (WFNA). Several of the more available fuels are: liquid methane (CH_4), gasoline, ammonia, and alcohol (CH_3OH or C_2H_5OH). Something better is sought than the common combination of alcohol and oxygen. This fact alone eliminates nitric acid as an oxidizer, regardless of its toxic and corrosive properties. Three promising propellant combinations remain: oxygen and hydrazine, oxygen and gasoline, and oxygen and ammonia. Oxygen is readily available and well known, as are ammonia and gasoline. Hydrazine or some of its close relatives is an excellent all around fuel, but at present it is expensive.

8.3 Auxiliary Power Supply

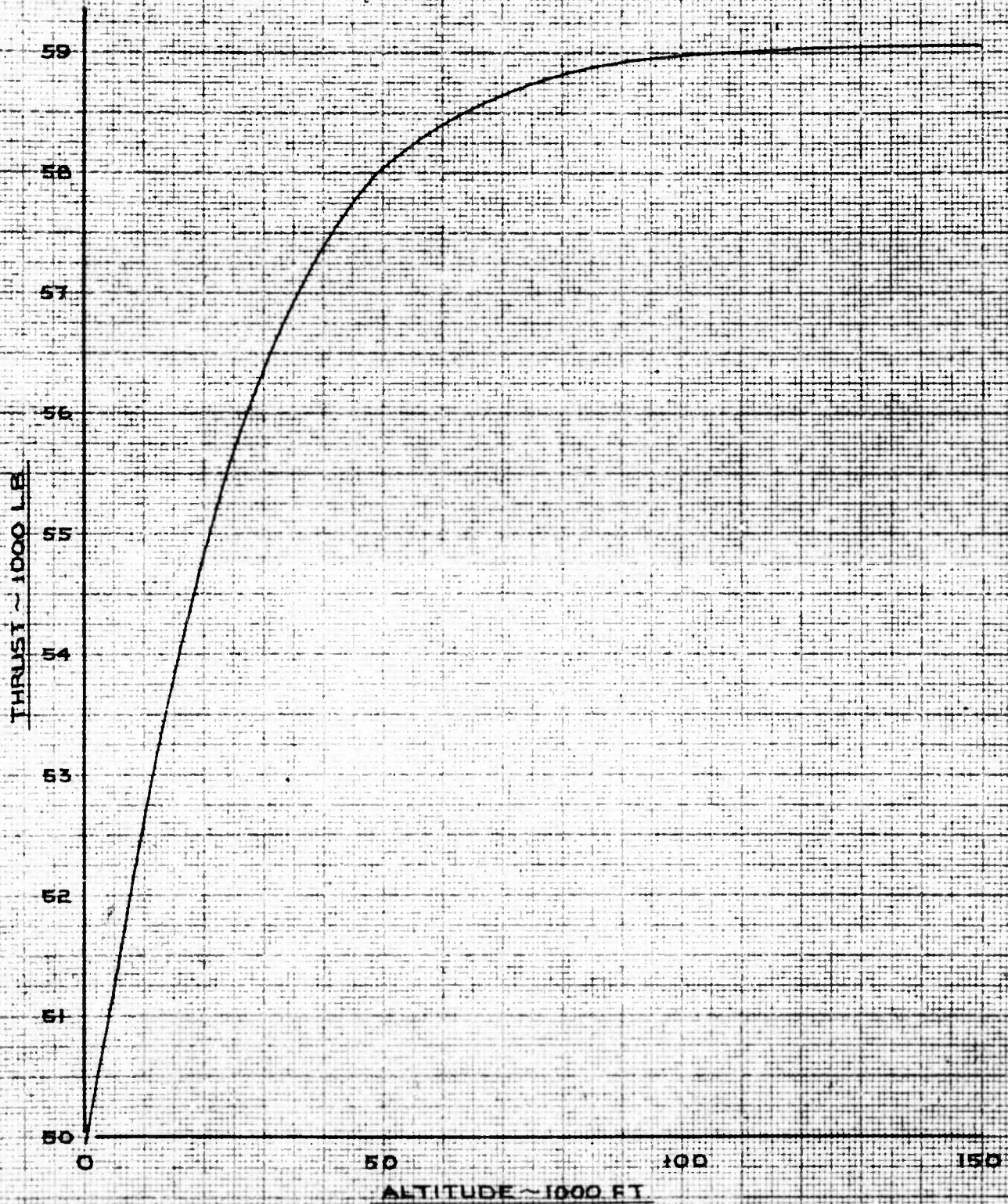
It is expected that a maximum of about 8 horsepower of electrical energy will be required to operate instruments, controls, radio, etc., with an average of about 3 horsepower. Investigation has shown that the lightest available power supply for the required duration will be an auxiliary generator set operating upon either hydrogen peroxide or ethylene oxide. Two companies are known to be developing and producing such auxiliary power units of about the proper size: the Walter Kidde Company and the American Machine and Foundry Company. According to both companies, a 10 horsepower hydrogen peroxide unit can be produced to provide 30 horsepower-minutes of 400-cycle current at a weight of about 56 pounds, including propellants. This weight is only a fraction of that for the equivalent energy in batteries.

TABLE IIIESTIMATED PERFORMANCE

Reaction Motors XLR 30-RM-2 Rocket Motor

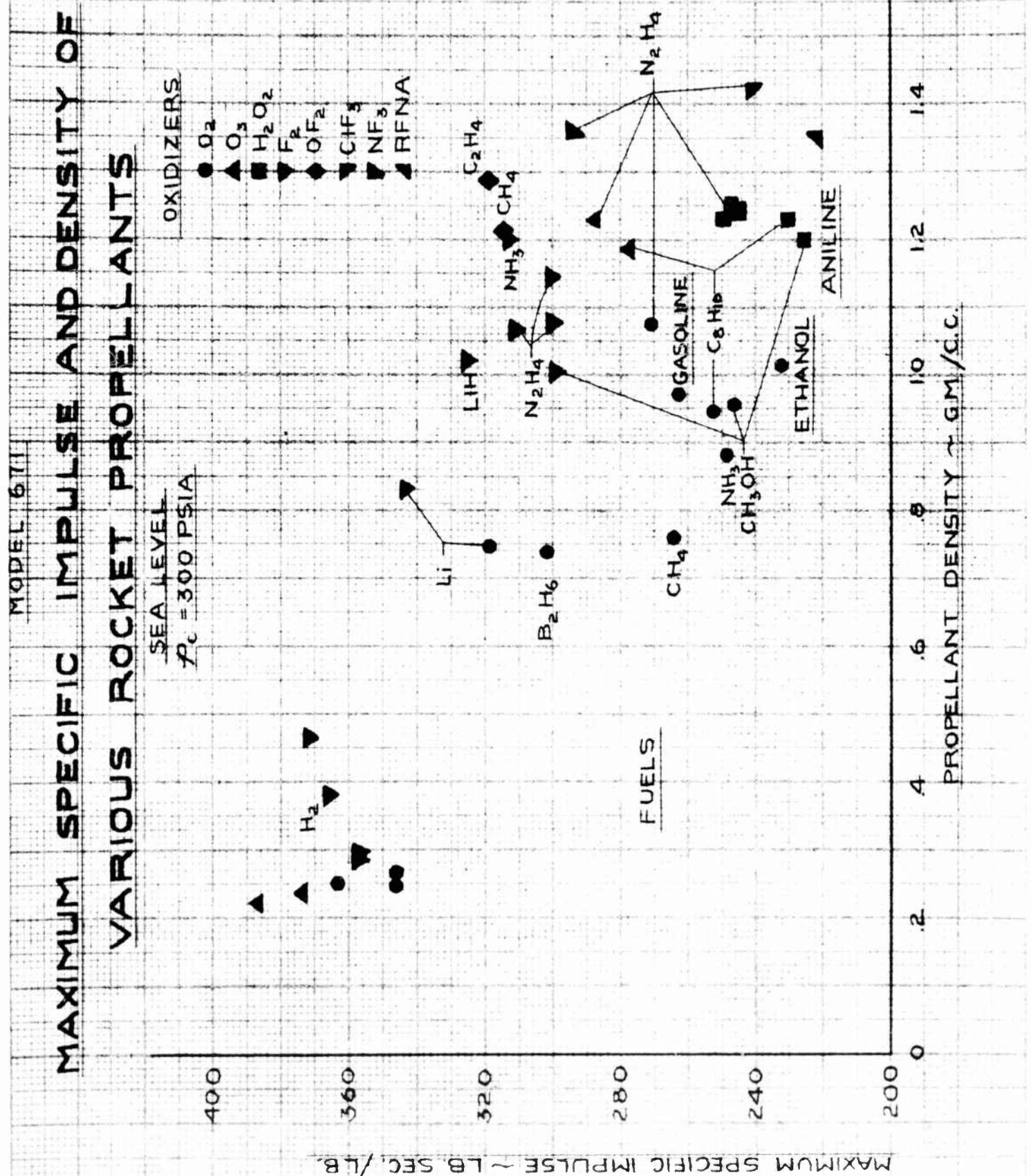
Thrust (static sea level)	(lb)	50,000
Specific Impulse	(sec)	245
Fuel Consumption	(lb/sec)	200
Weight	(lb)	395
Oxidizer		Oxygen
Fuel		Ammonia

MODEL 671

VARIATION OF THRUST WITH ALTITUDEREACTION MOTORS, XLR30-RM-2 ROCKET ENGINE

DOUGLAS AIRCRAFT COMPANY INC EL SEGUNDO PLANT EL SEGUNDO, CALIFORNIA

FORM 40-245-2 (4-49)



9.0 STRUCTURE

The structural arrangement of the airplane is conventional and presents no unusual problems in design. However, the extremely high temperatures and heat loads to which the airplane will be subjected during re-entry into the atmosphere require that careful attention be given to the optimum choice of structural materials, and to methods of insulating or cooling the surfaces. The boundary layer attains a maximum temperature of over 5000°F . Since this temperature is above the melting point of any structural material, various methods of preventing the structure from reaching this temperature were investigated.

Due to the comparatively low rate of convective heat transfer which occurs in the rear 80 percent of the wing and over practically the whole surface of the body, it is believed that a feasible method of construction for these components will be achieved by insulating the structure from the air stream. In using this method of construction the problem becomes one of choosing the proper amount of insulation to achieve the minimum combined weight of structure and insulation.

Since all structural materials suffer a loss of strength with increasing temperatures, as shown in Figure 28, the weight of the structural material will increase as the temperature of the structure is shown in Figure 29. Since the thickness of insulation required to prevent the rise of structure temperature increases with the difference between the desired structure temperature and the equilibrium temperature of the surface in this region (about 1800°F), the lower the structure temperature is held, the greater will be the weight of insulation. This relationship is also shown in Figure 29. The studies conducted indicate that the use of a titanium alloy of the nature of C-110M operating at a temperature of about 700°F would give the optimum combined structure plus insulation weight and the dashed line on Figure 29 is constructed on this basis. The use of an insulation having a different density-conductance relationship might alter this conclusion considerably. Since the spar webs and other internal structure would experience considerably less temperature rise, aluminum alloys will be used for these members. This will lower the thermal stresses, as the aluminum alloys have much greater coefficients of thermal expansion than the titanium alloys.

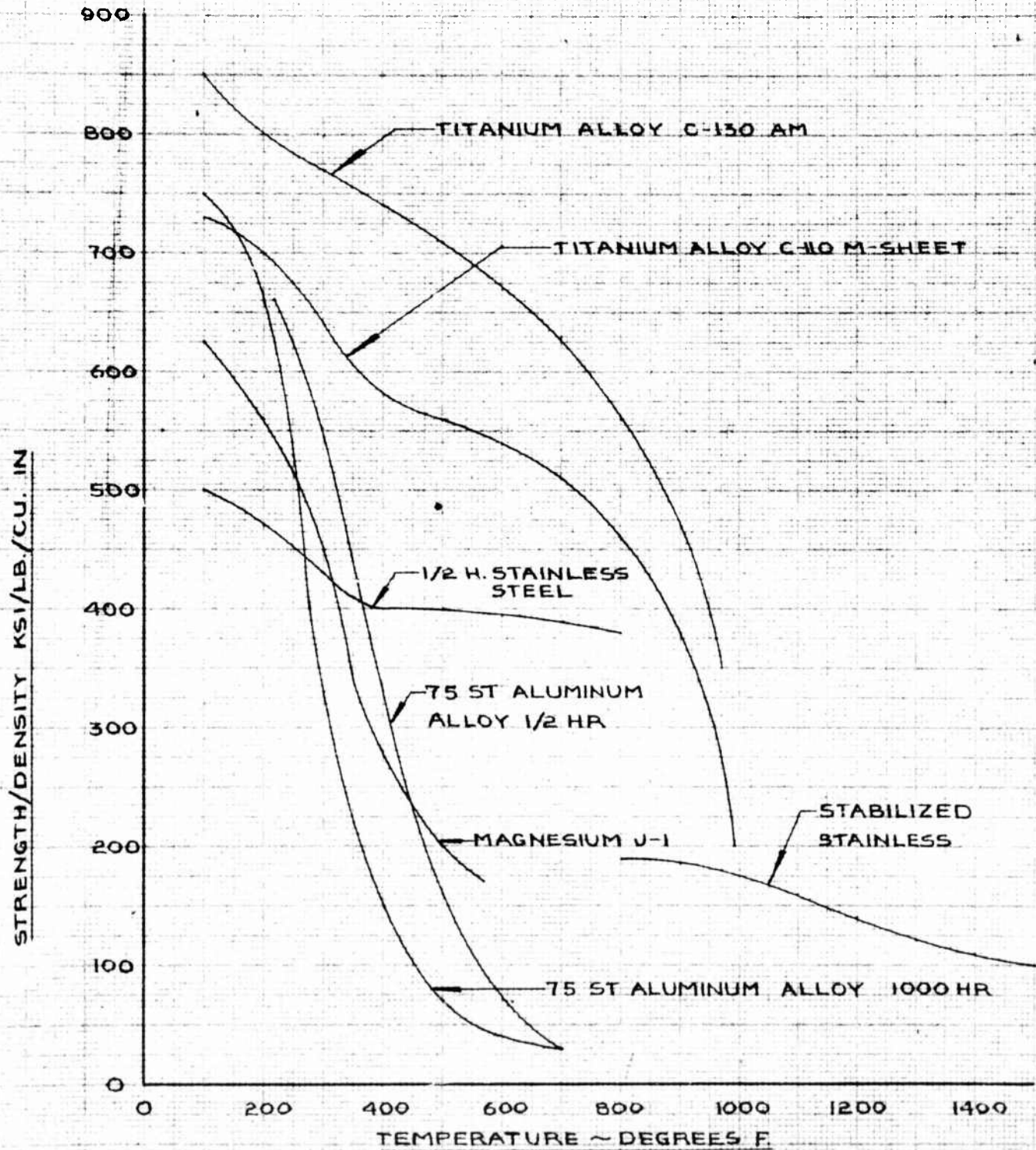
It should be emphasized that this type of construction depends upon the development of an insulating material suitable for exterior application to the surfaces. The material must possess good insulating properties, remain adhesive throughout the temperature range, and have about the same expansion characteristics

as the structure in order to avoid spalling. None of the insulation coatings now available are known to satisfy all of these requirements.

The maximum equilibrium temperature at the stagnation point is about 3300° F. As no suitable insulating material exists which could stand this temperature, the erosion of the air stream, and flexure of the structure, some other method of cooling must be used. One possible method of construction is to use a stainless steel non-structural nose section cooled to about 1500° F by the vaporization of water on the inside. During the ascent of the airplane and during all of the descent except about 35 seconds, the temperature at the stagnation point does not exceed 1500° F. The structure will be designed using the strength of the material at this temperature, and artificial cooling will be used only during the part of the flight when the temperature would tend to be higher than this figure. This cooling will be accomplished by spraying water at cockpit temperature into the leading edge section of each airfoil and conducting the resultant vapor along the span of the leading edges. The vapor will be at the pressure of the ambient outside air and will be exhausted at a temperature of approximately 1300° F. This gives a heat absorption of about 1625 BTU per pound of water. The total water required to keep all of the leading edge sections of the airfoils down to 1500° F is estimated to be about 45 pounds.

DOUGLAS AIRCRAFT COMPANY, INC. 11100 W. WASHINGTON AVENUE, CHICAGO, ILL. 60678

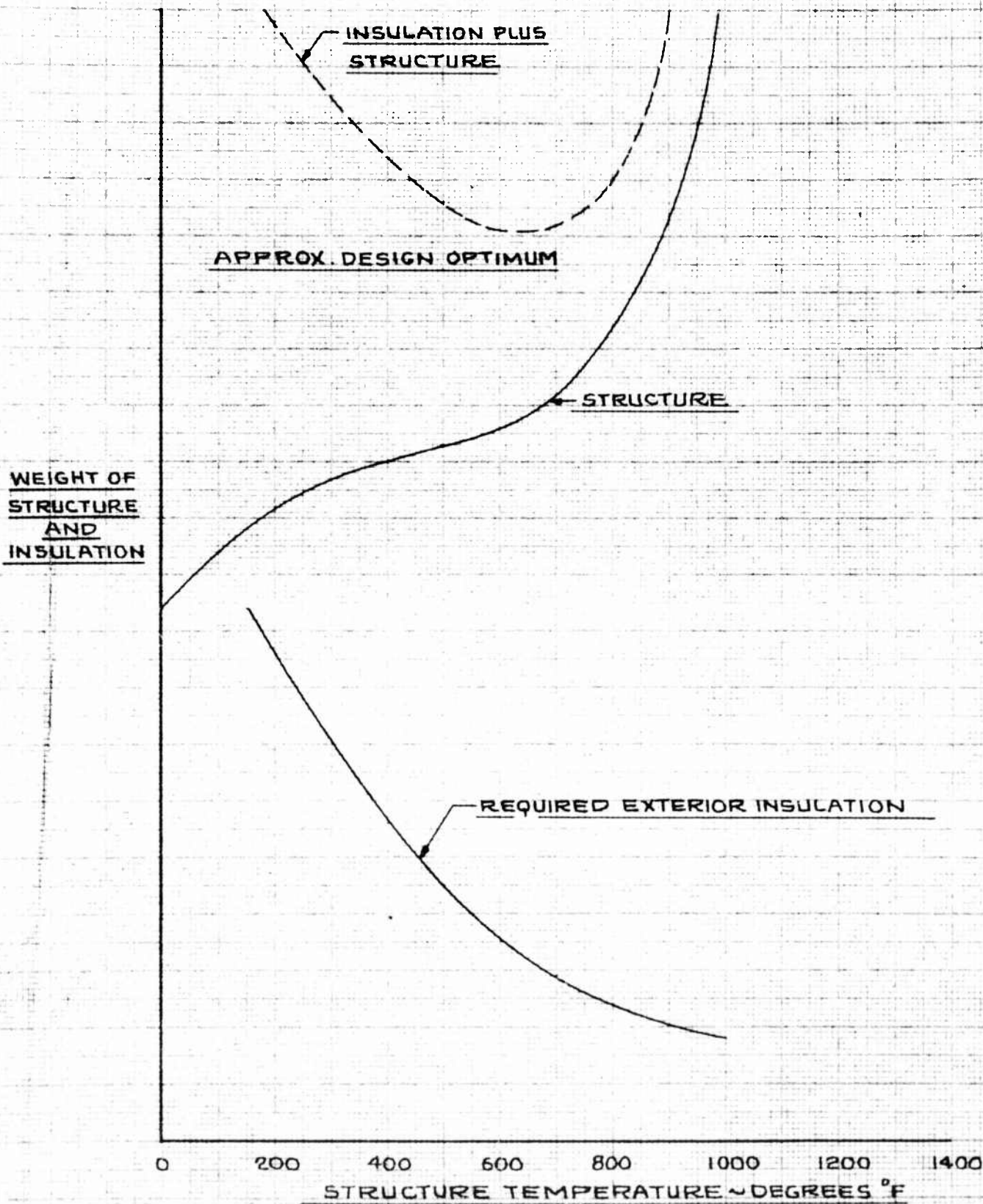
MODEL 671

EFFECT OF TEMPERATURE ON MATERIALS

DOUGLAS AIRCRAFT COMPANY, INC. 3100 WILSON AVENUE, SANTA MONICA, CALIF. 90405

MODEL 671

EFFECT OF TEMPERATURE ON WEIGHT



10.0 WEIGHT AND BALANCE SUMMARY

A summary of the airplane weight and balance is as follows:

TABLE IV
WEIGHT SUMMARY

Pilot	200 lb.	
Fuel-Rocket Engine	15,000	
Fuel-APU & Att. Control	35	
Water-Structure Cooling	45	
Test Equipment	150	
Pilot's Oxygen	18	
Useful Load		15,448 lb.
Weight Empty		6,752 lb.
Normal Gross Wt. - Gear Up (c.g. = 4% M.A.C.)		22,200 lb.
Gross Wt. Less Fuel - Gear Up (c.g. = 9% M.A.C.)		7,200 lb.

For details of the Weight Empty and the derivation of these weights see Reference 18.

Because of the extremely high temperatures the airplane is expected to encounter, it is necessary to replace the normal aluminum alloy skins with titanium and an insulating covering, which involves a weight penalty of 1100 lbs. in the Weight Empty. In addition, 45 lbs. of water is necessary for structure cooling as noted above.

A light weight electrical system is proposed consisting of a hydrogen peroxide turbine driving the alternator and sharing a fuel tank and reaction chamber with the attitude control rocket system.

Another feature of this proposal is the integral main propellant tanks which are pressurized initially by stored gas, and after starting by products of their own combustion in a secondary reaction chamber.

11.0 HUMAN ENVIRONMENT

The pilot will be exposed to an unprecedented combination of extreme environmental conditions. Accelerations will be severe in degree and duration. Atmospheric pressure and partial pressure of oxygen will be reduced to insignificant values. Temperatures of the aircraft outer skin will range from normal atmospheric to 1400°. Changes in sky and celestial brightness, celestial fragments, solar x-ray, cosmic and other short wave radiations will offer new hazards. On the other hand, it should be noted that the flight will be of short duration. The following discussions present the details of these problems together with an evaluation of their relative importance, and a description of methods by which their solution may be achieved in an effective manner.

11.1 Acceleration Loads

Longitudinal (front to back of pilot) accelerations encountered during launching in the 38° trajectory, aggregating 10 g in the course of a minute, are within tolerable limits. Positive (head to seat) accelerations of 7 to 8 g, lasting for 28 seconds during re-entry on this trajectory, are on the extreme borderline of tolerance with present accepted anti-g equipment. Higher angle trajectories would not be tolerable to the pilot in the light of presently known acceleration test data on human subjects.

Figure 30 shows a summation of existing acceleration load tolerance data for human pilots. Figures 31 and 32 show the resultant directions of the load factors acting on the pilot during the most critical periods of the take-off and re-entry with a standard type seat. Current studies at the Aviation Medical Acceleration Laboratory involving tolerance determinations in increments of 10 degree variations of seat axis may establish data which can be adapted advantageously to the design of a seat which will provide the optimum angle of application of acceleration force to the pilot's body in order to provide for greatest pilot tolerance during the critical re-entry accelerations.

Because the accelerations are marginal for the pilot, it is important that the aircraft be guided by automatic control during the high-acceleration portion of the flight path.

Sub-gravity accelerations and weightlessness will be more of a curiosity than a physiological threat. Present thinking and speculation (in the absence of experimental data) do not envision ulterior physiological changes incident to weightlessness other than effects upon neuromuscular requirements which are subject to adaptation. The possibility of disturbances of orientation by effects similar or opposite to the autogyral illusion cannot be answered until experimental studies establish their existence.

11.2 Cockpit Pressurization

The pressure in the cockpit will be maintained at a minimum pressure of 8 pounds per square inch absolute. The proposed cockpit pressure schedule is as follows:

- (a) From sea level to 5000 feet - no pressurization.
- (b) Above 5000 feet, a constant 5000 foot pressure altitude (12.23 pounds/square inch absolute) will be maintained until 8 pounds/square inch cockpit differential is achieved, at about 31,000 feet.
- (c) Above this 31,000 foot point, a constant 8 pounds/square inch cockpit differential will be maintained. Thus at altitudes where atmospheric pressure is essentially zero, the cockpit pressure altitude will be approximately 15,900 feet.

Control of the cockpit pressure will be by a pressure regulator exit valve very similar to those presently in use on existing military aircraft. One essential difference will be that the valve must be capable of sealing with zero leakage when the cockpit pressure is equal to or below the scheduled pressure level. An emergency cockpit pressure relief valve would also be provided to prevent an increase in cockpit pressure beyond structural limits, as might be occasioned by a malfunction of some part of the pressure source or control mechanism.

The source of pressure for the cockpit will be a liquid oxygen system. This system will also supply cockpit cooling and breathing oxygen. A complete description of this equipment will be given in the following paragraphs on cockpit air conditioning.

In order to conserve oxygen and provide adequate cockpit pressure for the desired length of time, it will be necessary that extreme care be taken in designing a leaktight cockpit structure and leaktight seals. Experience presently being acquired in the extreme low leakage cockpits of the Douglas A4D-1 and F4D-2 airplanes can be applied to this project.

To protect the pilot in an emergency arising from failure of the main source of oxygen or rupture of the cockpit walls, a pilot pressure suit complete with emergency gaseous oxygen supply will be furnished. The pressure suit will be of the ventilated type presently being perfected for conventional military aircraft. Emergency pressure differential in the suit will be approximately 3 pounds/square inch.

11.3 Cockpit Air Conditioning

11.3.1 Basic Air Conditioning System

After a study of several possible systems, it was decided that the best system for this aircraft would be one using stored liquid oxygen as the basic coolant and source of cabin pressure. A schematic diagram of this system is shown in Figure 33. This system is the lightest and simplest of several considered. Operation of the system centers around a liquid oxygen container similar to those presently in use on military aircraft for breathing oxygen. By proper design, heat transfer to the liquid oxygen in the storage container will keep the container charged to a pressure of about 85 pounds/square inch absolute pressure. From this pressurized oxygen container, a controlled amount of liquid oxygen will flow into the heat exchanger. Here the oxygen will cool the air being recirculated from the cockpit. The oxygen, by this time expanded into the gaseous phase, will then be jetted into the circulating cockpit air stream, thus furnishing the motive power for the recirculation process. The temperature control system which modulates the flow of cooling oxygen to meet the cockpit and pilot pressure suit requirements will be powered completely by oxygen pressure. Humidity control will be provided by circulating the cockpit air through a canister of Silica Gel, Sova Beads, or other dessicant material. For the short flights contemplated, odor and fume removal, and supplementary heating systems are not believed to be necessary.

11.3.2 Windshield Defrosting

It is believed that if the pilot pressure suit is vented overboard so that moisture from the pilot's body does not enter the cockpit atmosphere, and if control of humidity is maintained by the dessicant material noted in the above paragraph, interior windshield defrosting will not be required.

11.3.3 Cockpit Thermal Insulation

To provide for a minimum amount of cockpit temperature variation and a minimum consumption of cooling oxygen, it will be necessary to use a highly effective thermal insulation on the cockpit walls. As has been mentioned previously in this report, an 0.025-inch thickness of insulation material on the outer surface of the structure will reduce the maximum structure temperature from about 1400° F to about 750° F. To further reduce the temperature of surfaces adjacent to the pilot, two inches of aircraft quality, high temperature, fiber glass batt or equivalent will be used on the inner side of all cockpit surfaces. In addition to the batt, thin layers of highly polished aluminum foil will be used under the batt to minimize heat radiation to or from the cockpit skin. According to transient heat transfer studies, the inner surface (surface exposed to pilot) of this proposed insulation should not

exceed a maximum temperature of about 120°F during the time required to execute the planned flight path of the aircraft. This maximum would occur on the re-entry portion of the flight and therefore will not be of long duration.

11.3.4 Windshield Construction

In order to resist the high temperatures involved, especially during re-entry, it will be necessary that the windshield be constructed of transparent quartz. Heat transfer studies indicate that two one-half inch layers of quartz with a one-quarter inch vented air gap between would be sufficient to drop the inner windshield surfaces below 200°F. As will be mentioned again in a later paragraph, a thin treated glass will be used on the pilot's side of the quartz layers to reduce ultra-violet and other harmful radiations entering through these transparencies.

11.4 Oxygen Supply

Normal oxygen supply would be from the liquid oxygen container used also for cabin pressurization and air conditioning as described above. It was previously noted that a small emergency bottle of gaseous oxygen would also be provided.

11.5 Radiation

Much information concerning the intensities and biological effects of a wide range of solar and cosmic radiations is becoming available. Even though many of our best informed physicists and biologists now tend to be optimistic, no one can categorically state that serious biological radiation damage will not result from extended flight in free space. Fortunately for the problem of the aircraft at hand, the exposure time is no longer than a few minutes per flight. Proper precautions to prevent any one pilot from making too many successive flights in a weeks or months time interval should be taken, as repeated exposures with short time lapses could cause additive effects.

11.5.1 Solar Ultra-Violet Radiation

Solar ultra-violet radiation without the atmosphere's protective influences would in itself be a primary hazard except for the fortunate fact that it can be easily excluded even by relatively thin layers of protective material. As noted above, the windshield of the airplane will be of quartz, which is relatively transparent to ultra-violet radiation. For this reason it will be necessary to add a thin treated glass layer on the inner side of the quartz layers to shield the pilot from this particular radiation.

11.5.2 Solar X-Rays

Present indications are that solar x-rays will present a relatively unimportant problem. This matter will be further studied and if necessary additional protective elements will be added to the transparent ultra-violet shield.

11.5.3 Cosmic Rays

The above discussion leaves the cosmic particles and other heavy ionizing radiations as the major worries. However, as mentioned above, it will be possible to control the time of each pilot's exposure to a pre-determined number of minutes per week or month, as may be considered optimum by careful observation. During any one flight (of approximately ten minutes duration), damage to the pilot's body cells which are replaceable and to nerve cells which are not replaceable should be small. According to various radiation authorities (References 19 & 20), the number of nerve cells destroyed by heavy particle penetration would be insignificant for a period of exposure much longer than the few minutes which would occur in the projected flight of this aircraft. Figure 34 shows values of permissible radiation doses and expected intensities versus altitude. The data shown in this figure are among the best now available, but do not cover all possible effects of heavy cosmic particles and their secondary reaction.

No known practical approach toward shielding against these particles is yet available. Present test data show that ionization from certain heavy particles is still increasing after penetrating to the center of a two foot diameter lead sphere. Insofar as any known techniques are concerned, the possibility of deflecting these particles by high energy magnetic fields appears to be even less practical than the extremely heavy mass shielding. Partial shielding of the pilot either intentionally or unintentionally should be avoided insofar as is possible. Partial shielding results in secondary radiations due to heavy cosmic particle break-up. These secondary radiations can be more dangerous, by way of covering more area in the pilot's body, than the primary radiations. Extremely heavy atomic weight metals such as lead should be avoided, as present knowledge indicates they will increase radiation danger rather than decrease it. Light atomic weight materials, such as hydrogen, cause the least break-up of heavy particles and thus may prove best for cosmic radiation shielding. The design of hydrogen radiation shields has not been perfected either by experiment or by actual usage. Further tests by agencies studying the radiation shielding problem plus actual flight data on radiation at high altitude during short time exposures will accelerate the integration of such shields into extreme high altitude aircraft. However, at the present state of knowledge, shielding is considered to be of highly questionable value for the subject aircraft.

11.6 Meteor Collision

It has been calculated that an encounter with a meteorite with sufficient energy to penetrate a 1/16 inch stainless steel skin may be anticipated to occur once in 2,000,000 flights of this type aircraft. The chance of damage or injury from this source is infinitesimal and does not deserve design consideration. The best of current data on meteorites is attached as Figure 35.

11.7 Cockpit Arrangement

A preliminary layout of the cockpit area has been made, as shown in Figure 36. The general arrangement has been based on the assumption that many preliminary flights will be made under moderate flight conditions, in which the pilot will be able to control the aircraft without great difficulty. Requirements of flights to maximum altitude will not further complicate this simple cockpit control and instrument arrangement, since these flights may be controlled automatically or from a remote station. The number and type of instruments have not been definitely established but current thinking is that normal navigation instruments will be provided together with such special instruments as are required by the type of flight being made and by the information needed by the pilot. Besides the conventional aerodynamic and engine controls, switches to connect and disconnect the automatic flight control mechanism may be provided. It is assumed that the majority of flight data will be obtained by ground station equipment such as radar. No suitable equipment for sensing all of the test information desired from the maximum altitudes noted is known to be in existence at this time.

11.8 Emergency Escape Provisions

Because of the high speed and high altitude performance of the aircraft involved, it is believed that ordinary bailout or ejection seat procedures will be of little or no use to the pilot for emergency escape. Current thinking on ejection seats is that they are suitable up to a Mach number of approximately one at sea level, with somewhat higher speeds being safe at the higher altitudes. However, in view of the high temperatures which are anticipated during the re-entry problem from extreme high altitude, it would be impossible for an unprotected pilot to survive by either ejection chute bailout or standard ejection seat bailout procedures.

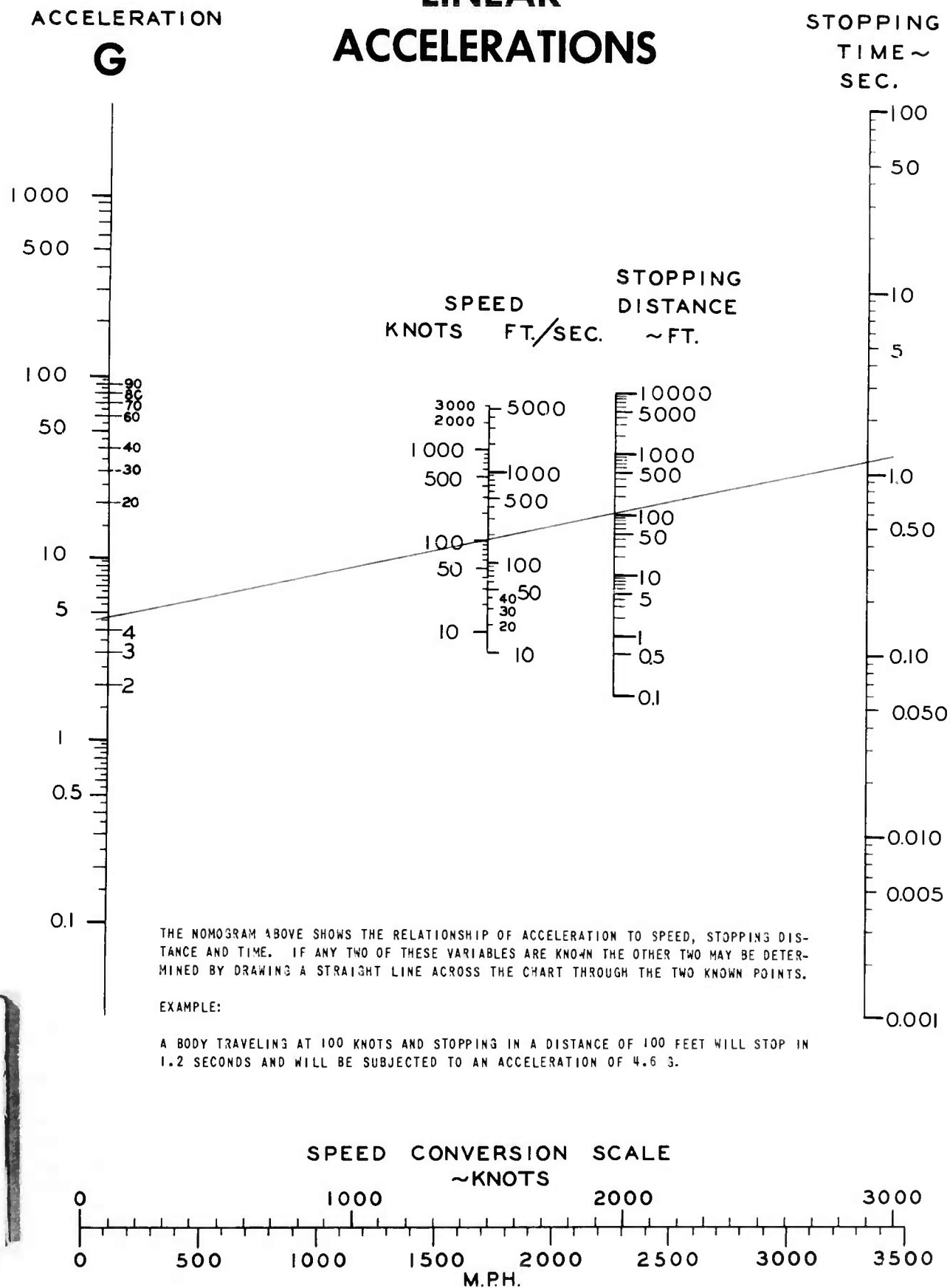
In addition to the high speed and temperature problems, there is the problem of environmental pressure. While the pilot may wear a pressure suit for emergency in the event of cabin pressure failure, it is very doubtful that sufficient pressurizing equipment could be carried by the pilot during standard bailout or ejection procedures to sustain suit pressure from the maximum altitude to a safety zone within the earth's lower atmosphere.

While the problem of escape may not be as severe during the preliminary exploratory flights, it is still considered that an ejection seat or other ordinary bailout techniques will be inadequate in view of the problem of high speeds and high altitude. Therefore it is believed that if any emergency escape system is to be provided, the ejectable cockpit type of system would offer the optimum protection for a reasonable weight increase.

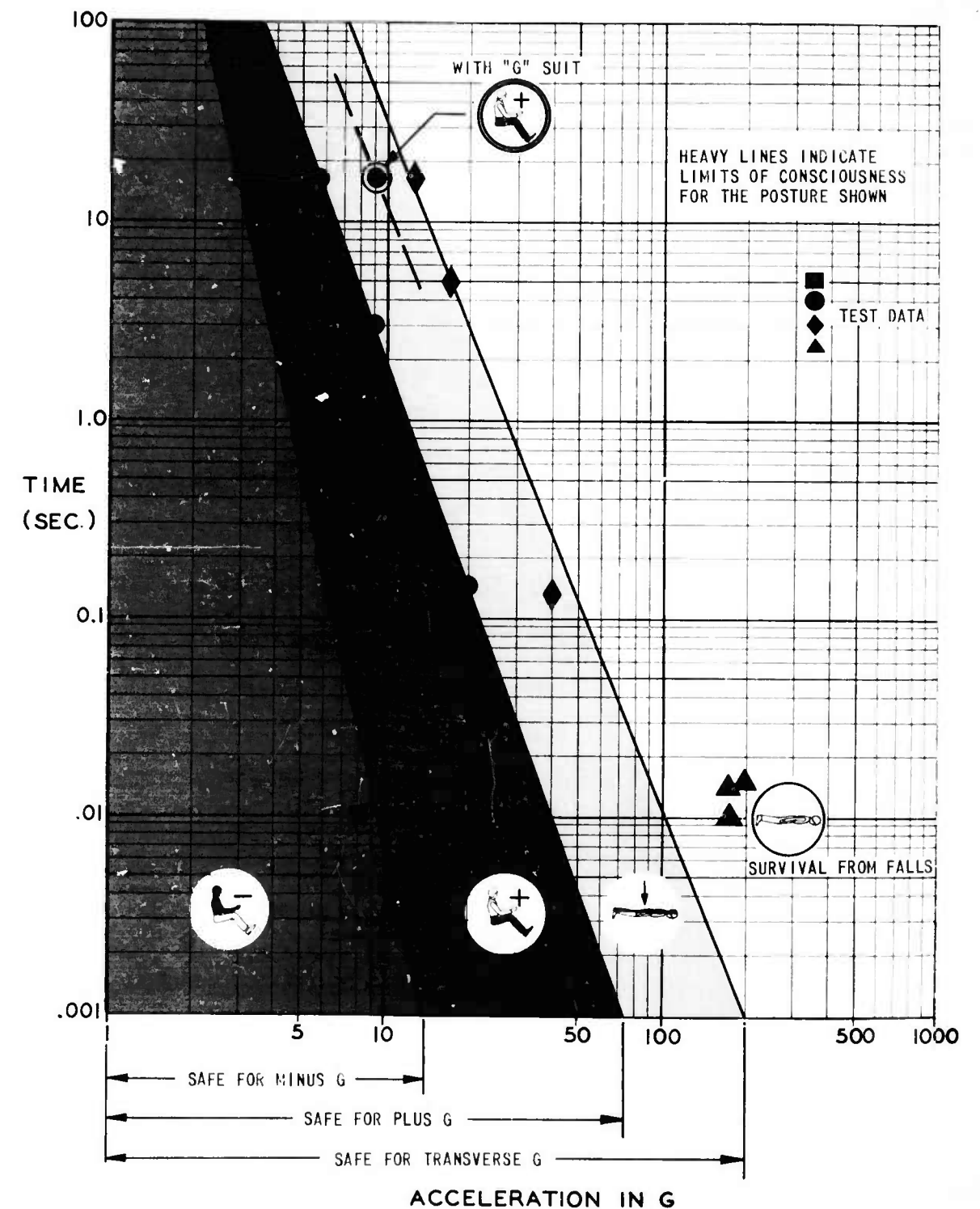
A detailed analysis of the ejection seat installation has not been made. On the basis of current installations, however, it would appear that a weight penalty of 50 to 75 pounds would be the minimum for such an installation. An ejection seat capsule is not considered in this problem because the space required would make it impossible to enclose the capsule within an airplane configuration which would meet the weight requirements of the project. The weight penalty of the cockpit capsule installation similar to that shown in Figure 37 is estimated to be about 130 pounds. This includes the additional structure, the parachutes, disconnects and the rocket unit for forcibly separating the cockpit from the remainder of the aircraft. It is recommended that this type of emergency escape provision be considered, on the basis that the majority of flights will be in the preliminary stages of the project where the escape capsule would offer a reasonable chance of survival. For the ultimate flight conditions, it appears doubtful that any of the present escape methods offer very much chance of survival.

ACCELERATION AND THE HUMAN BODY

LINEAR ACCELERATIONS



HUMAN TOLERANCE



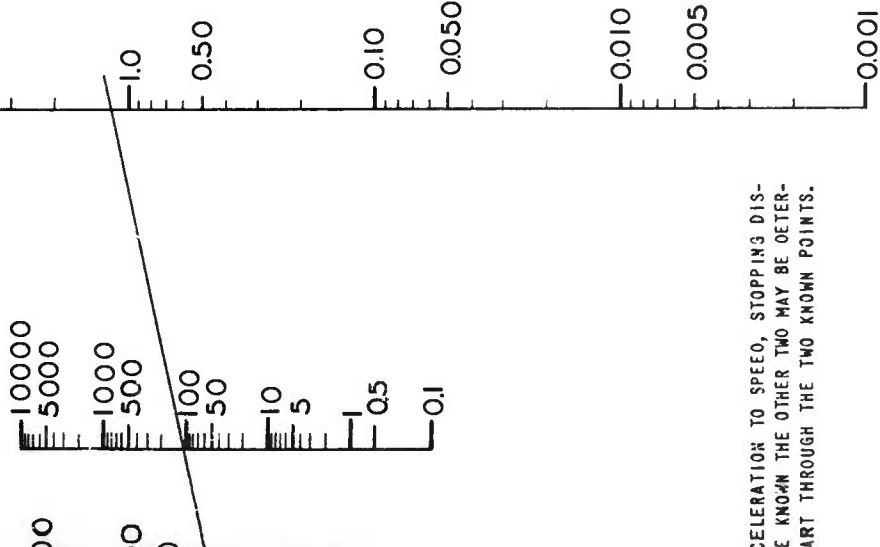
THE CHART ABOVE INDICATES THE TOLERANCE OF THE AVERAGE MAN TO ACCELERATIONS FOR A GIVEN PERIOD OF TIME. THE BLUE AREA INDICATES TOLERANCE TO NEGATIVE ACCELERATIONS; THE BLUE PLUS THE GREEN AREAS, TOLERANCE TO POSITIVE ACCELERATIONS; THE BLUE PLUS THE GREEN PLUS THE YELLOW, TOLERANCE TO TRANSVERSE ACCELERATIONS. FOR EXAMPLE THE AVERAGE MAN CAN WITHSTAND SAFELY AN ACCELERATION OF 10 G FOR A PERIOD OF ONE SECOND IF IT IS APPLIED IN A POSITIVE OR A TRANSVERSE DIRECTION, BUT WOULD FIND IT DANGEROUS IF APPLIED IN THE NEGATIVE DIRECTION.

ACCELERATION AND THE HUMAN BODY

RELATIONS

STOPPING
TIME ~
SEC.

STOPPING
DISTANCE
~FT.



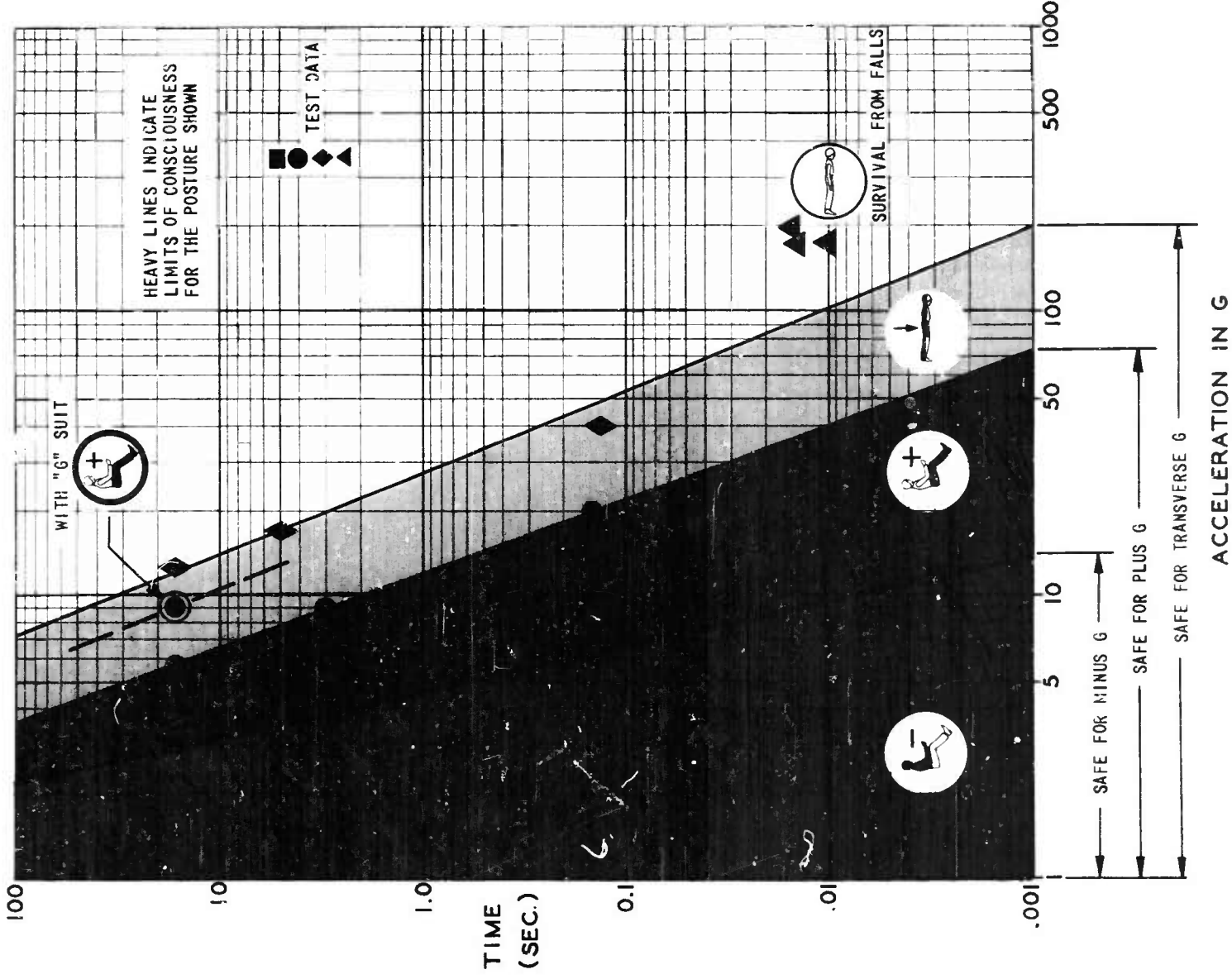
ACCELERATION TO SPEED, STOPPING DIS-
TANCE KNOWN THE OTHER TWO MAY BE DETER-
MINED THROUGH THE TWO KNOWN POINTS.

DISTANCE OF 100 FEET WILL STOP IN
PERIOD OF 4.6 SEC.

ON SCALE

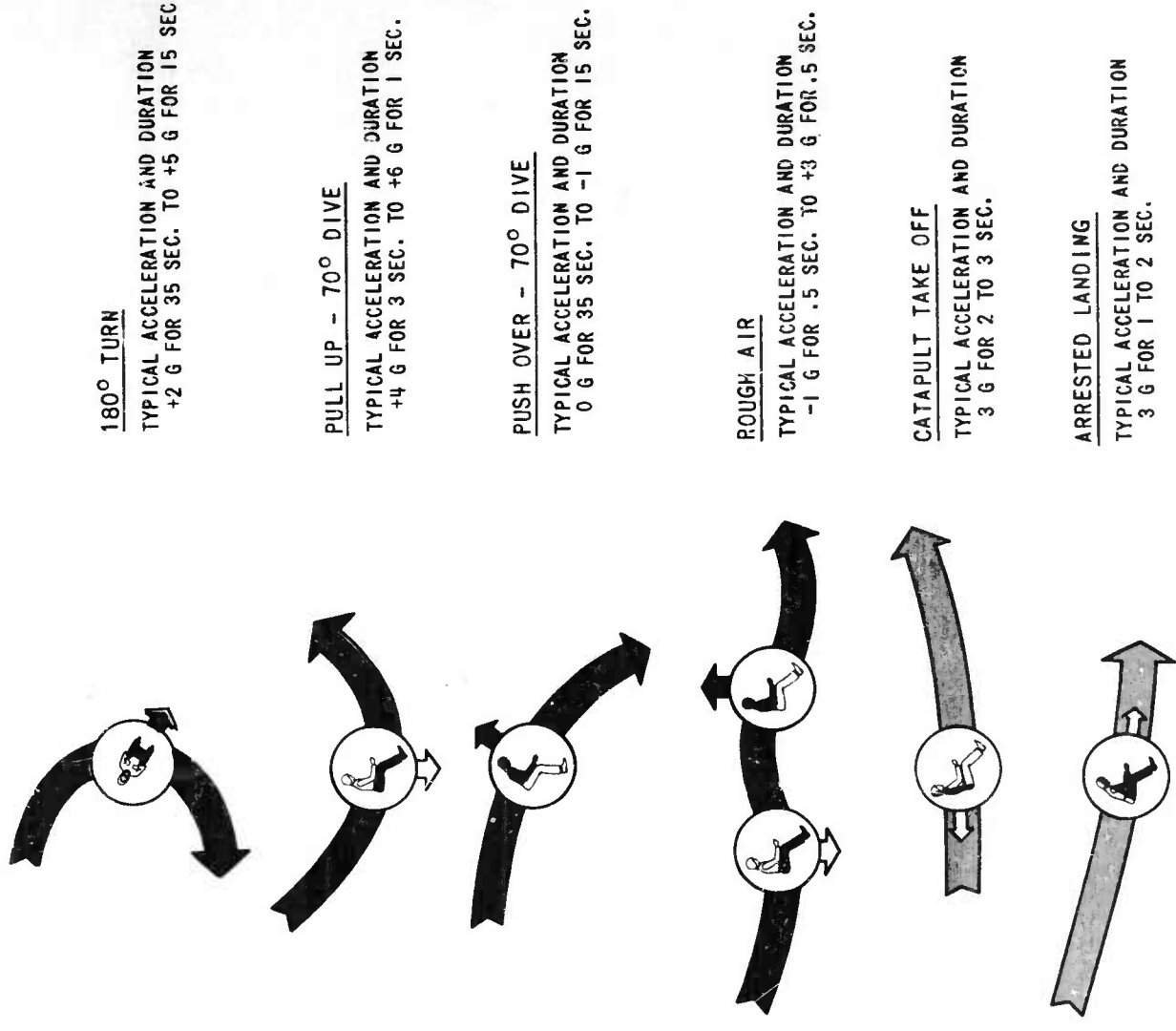


HUMAN TOLERANCE



THE CHART ABOVE INDICATES THE TOLERANCE OF THE AVERAGE MAN TO ACCELERATIONS FOR A GIVEN PERIOD OF TIME. THE BLUE AREA INDICATES TOLERANCE TO NEGATIVE ACCELERATIONS; THE BLUE PLUS THE GREEN AREAS, TOLERANCE TO POSITIVE ACCELERATIONS; THE BLUE PLUS THE GREEN PLUS THE YELLOW, TOLERANCE TO TRANSVERSE ACCELERATIONS. FOR EXAMPLE THE AVERAGE MAN CAN WITHSTAND SAFELY AN ACCELERATION OF 10 G FOR A PERIOD OF ONE SECOND IF IT IS APPLIED IN A POSITIVE OR A TRANSVERSE DIRECTION, BUT WOULD FIND IT DANGEROUS IF APPLIED IN THE NEGATIVE DIRECTION.

TYPICAL MILITARY MANEUVERING ACCELERATIONS



PREPARED BY:

DOUGLAS AIRCRAFT COMPANY, INC.

ENGINEERING DEPARTMENT

EL SEGUNDO DIVISION - EL SEGUNDO, CALIFORNIA

1951



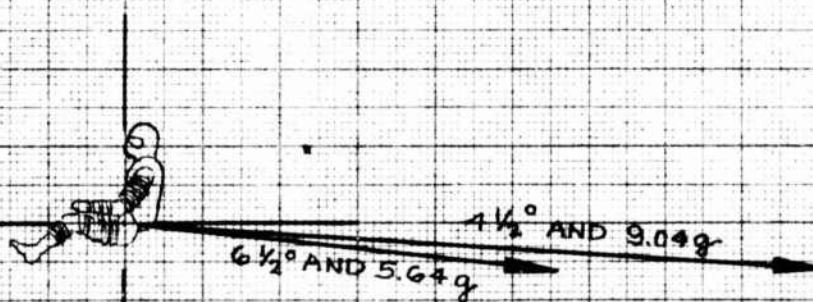
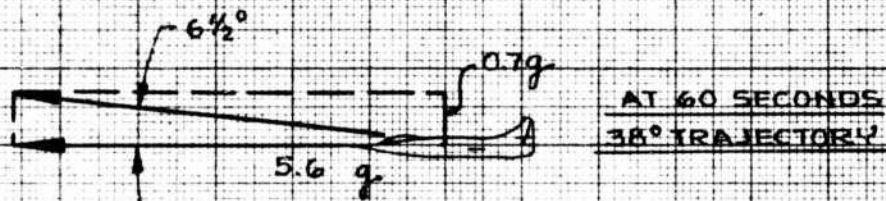
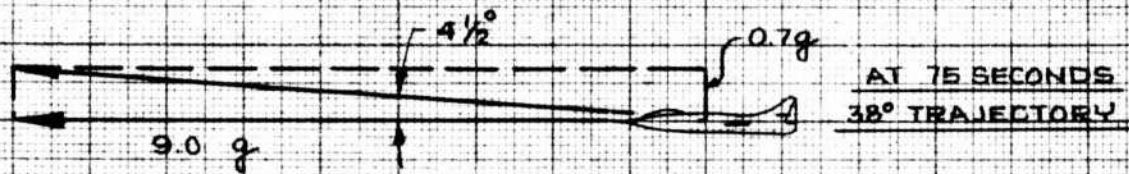
DOUGLAS AIRCRAFT COMPANY, INC., EL SEGUNDO DIVISION, EL SEGUNDO, CALIFORNIA

FORM 30-250-1 (9-51)

MODEL 671

LOAD FACTORS DURING TAKE-OFF

38 DEGREE TRAJECTORY



FORCES ON PILOT

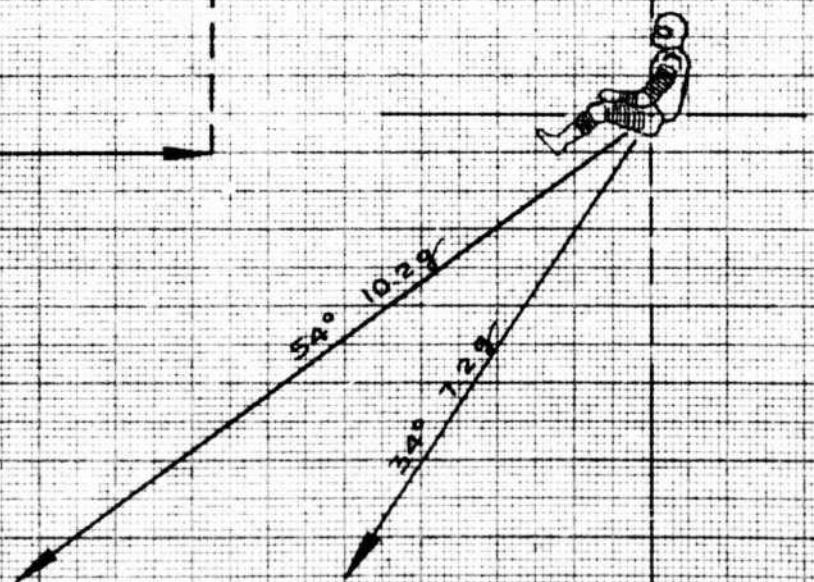
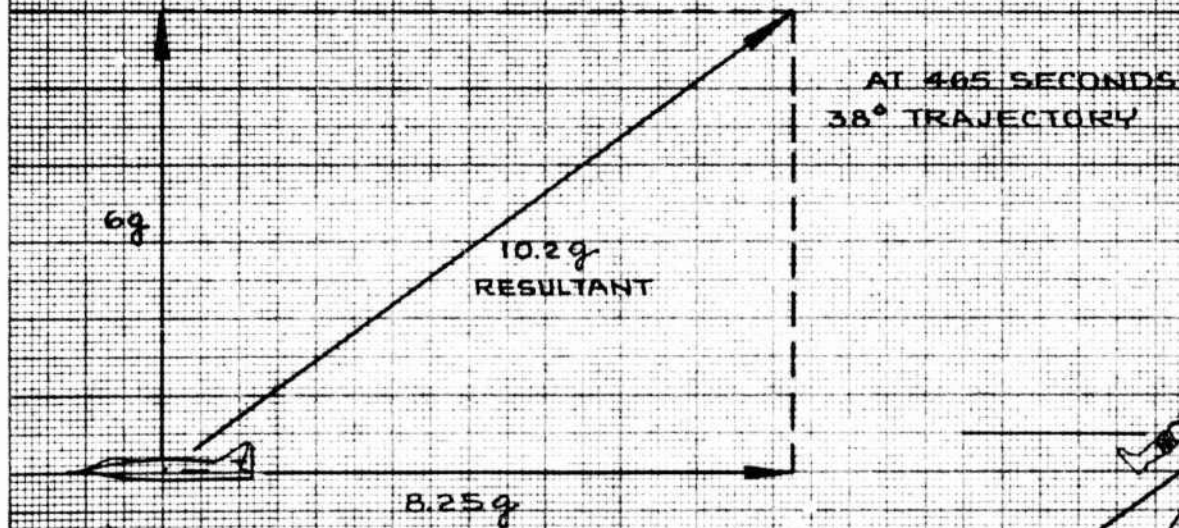
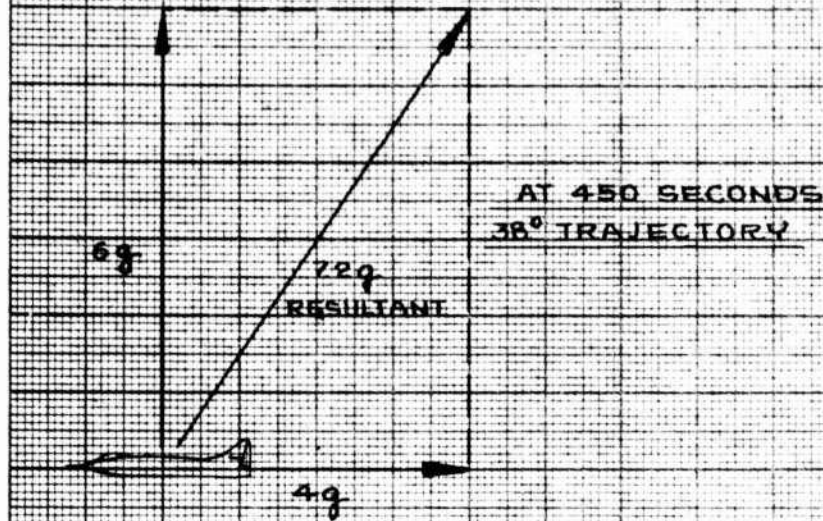
DOUGLAS AIRCRAFT COMPANY, INC., EL SEGUNDO DIVISION, EL SEGUNDO, CALIFORNIA

FORM 30-250-1 (9-51)

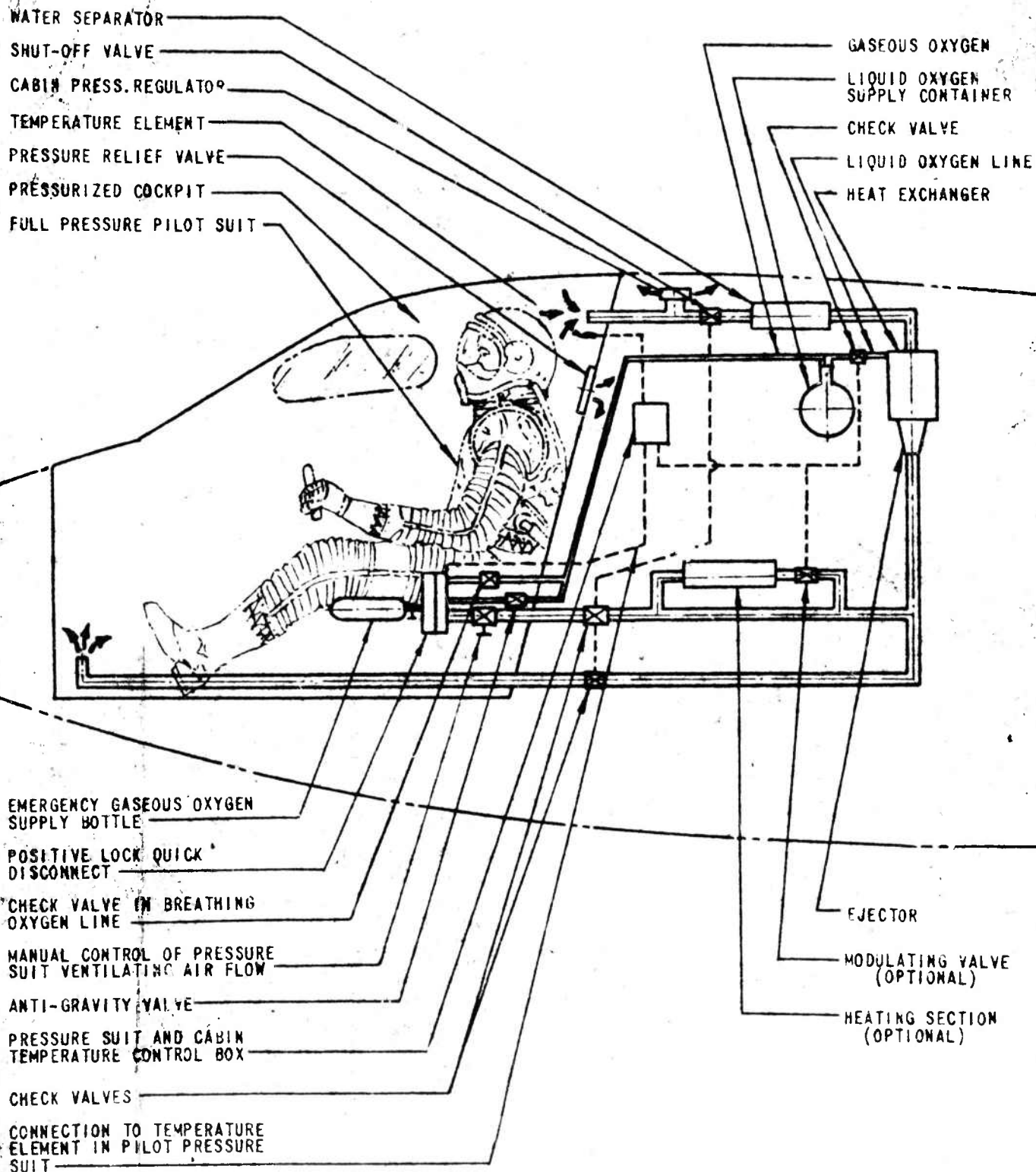
MODEL 671

LOAD FACTORS DURING RE-ENTRY

38 DEGREE TRAJECTORY



FORCES ON PILOT

AIR CONDITIONING SYSTEM

DOUGLAS AIRCRAFT COMPANY, INC. EL SEGUNDO DIVISION EL SEGUNDO, CALIFORNIA

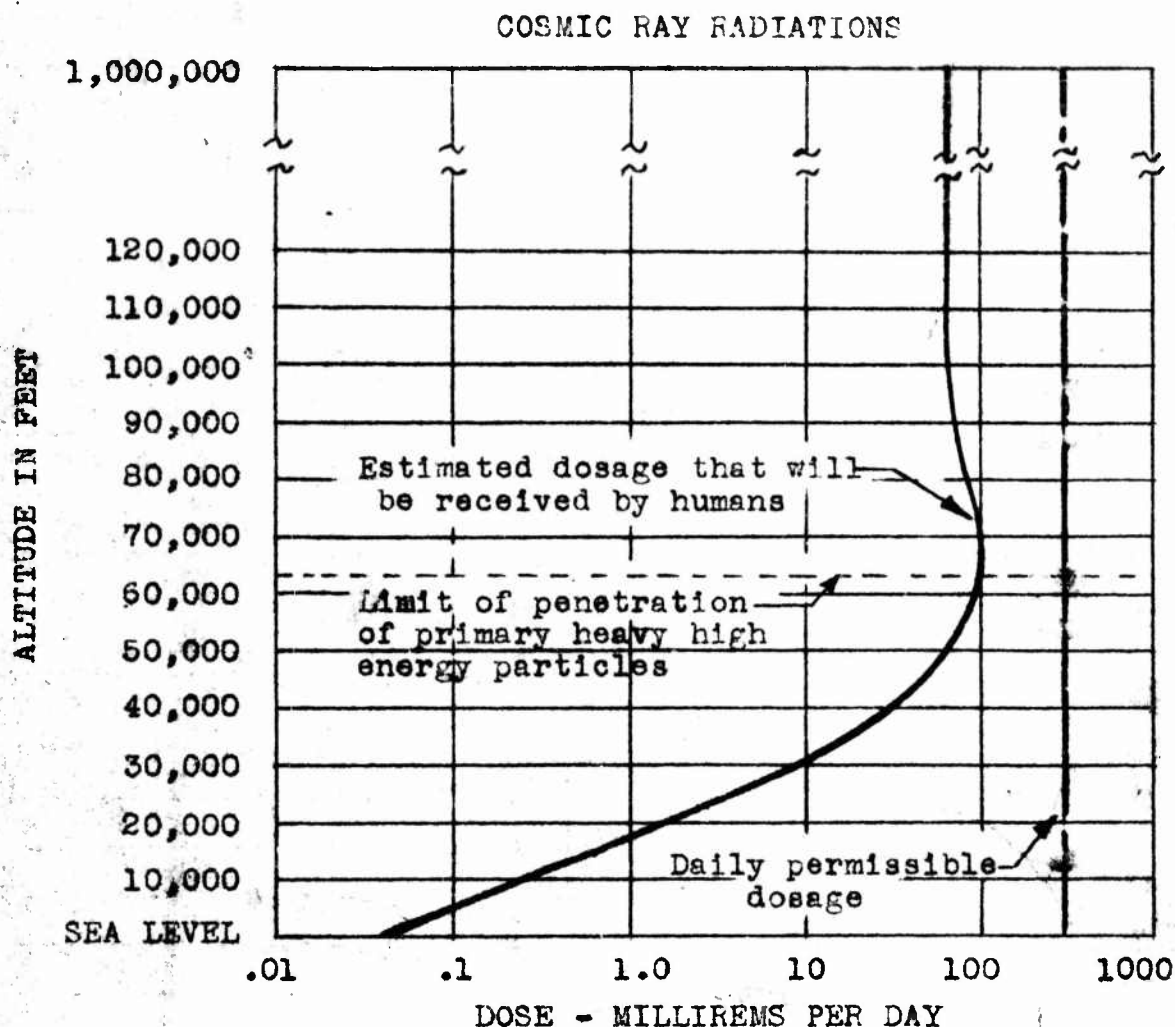


FIGURE 34 PERMISSIBLE RADIATION DOSAGE VERSUS ESTIMATED DOSE-AGE WHICH WILL BE ENCOUNTERED AT HIGH ALTITUDES

In the above chart the dosage unit "Millirem" is used. This is 1/1000 REM or "Roentgen Equivalent Man." The REM unit denotes the dose of 200 KV x-rays that would be necessary to produce quantitatively the same biological effect as a given dose of other radiation.

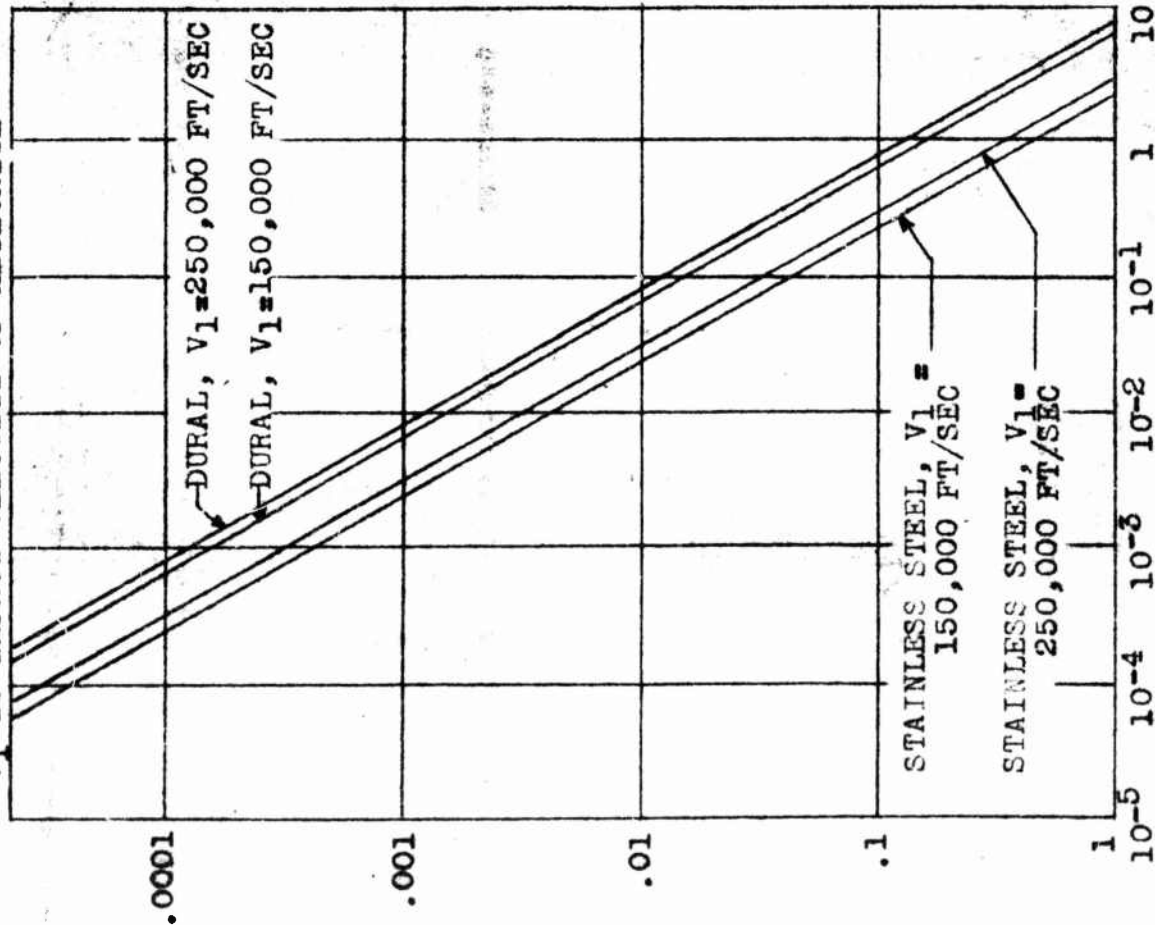
Daily permissible dose of 300 Millirems per day was set by U.S. National Committee on Radiation in 1949.

The above curve on radiation was estimated from data taken from References 19 and 20.

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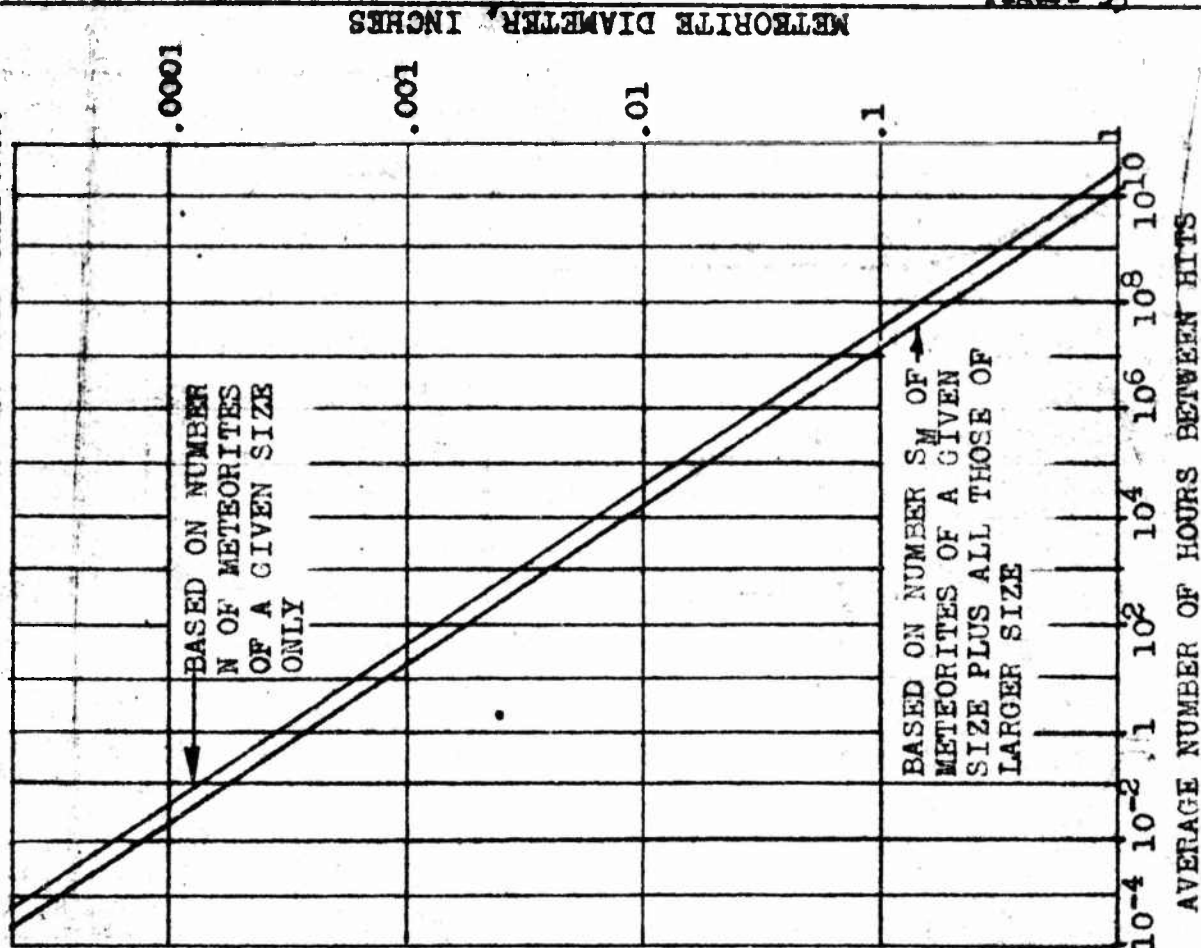
PENETRATION OF METAL PLATE BY METEORITES

BASED ON ASSUMPTION THAT METEORITE
IS A SPHERE OF DENSITY 3.4 GRAMS/CM³
V₁ IS IMPACT VELOCITY OF METEORITE



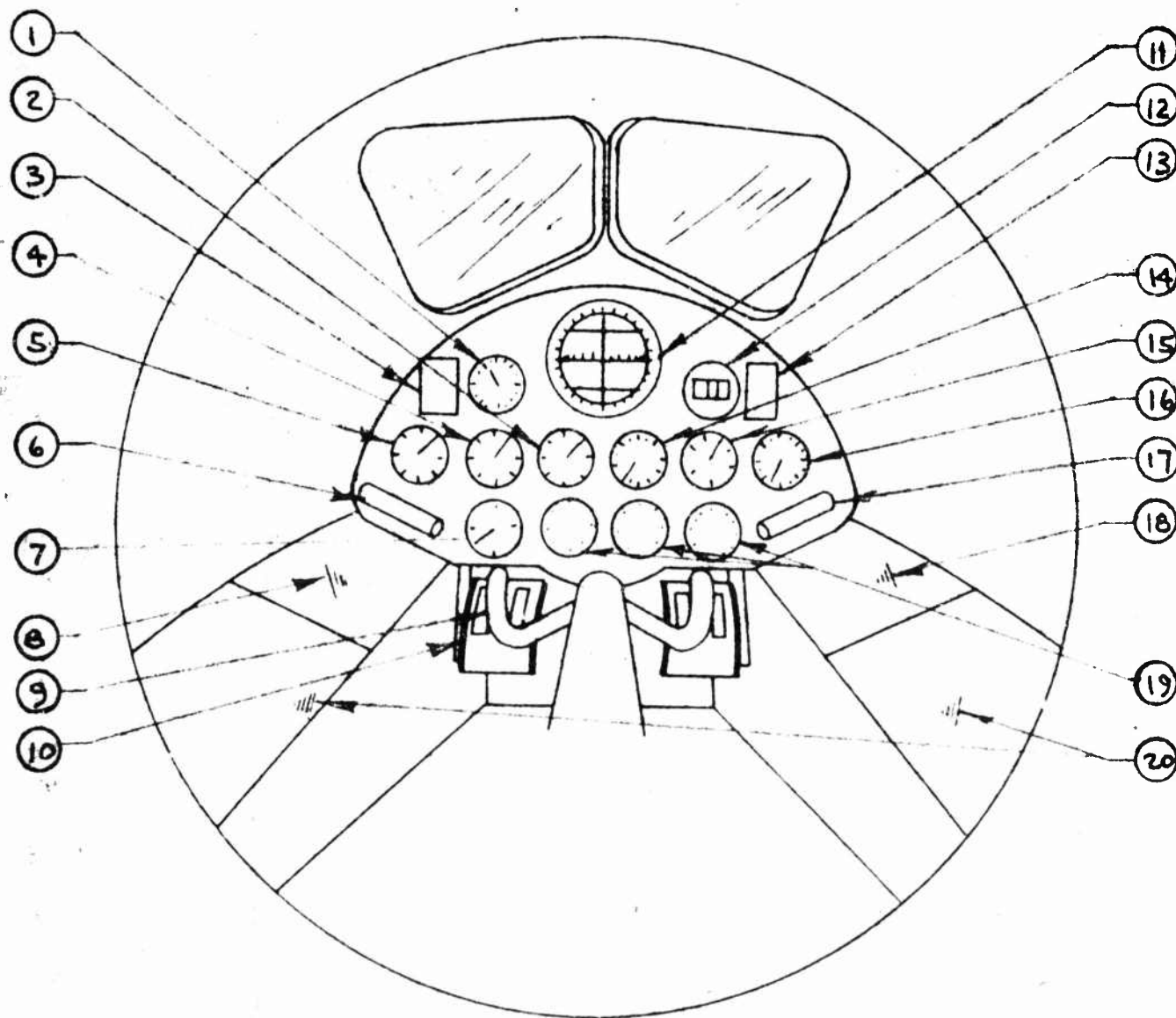
PROBABILITY OF A METEORITE HIT

EXPOSED AREA = 1000 SQUARE FEET.
DISTRIBUTION OF METEORITES ASSUMED UNIFORM.

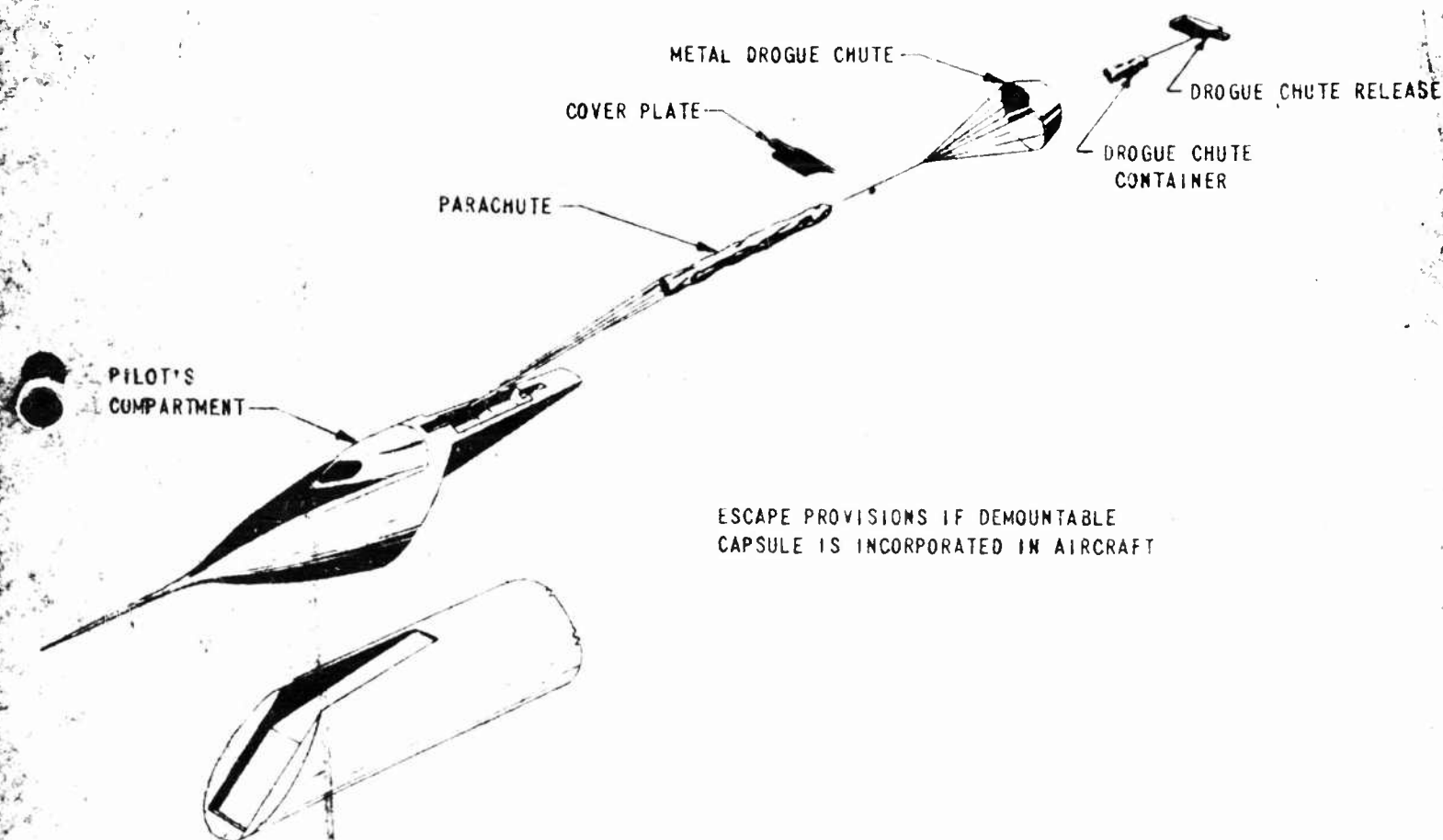


COCKPIT ARRANGEMENT

- | | |
|--------------------------------------|--|
| 1. Altimeter | 11. Composite Flight Indicator |
| 2. Airspeed Indicator | 12. "Rockets Seconds Remaining" Indicator |
| 3. Take-Off Check List | 13. Landing Check List |
| 4. Cabin Pressure Supply Indicator | 14. Mach Meter |
| 5. Cabin Altimeter | 15. Engine Inlet Manifold Pressure Indicator |
| 6. Jettisoning Handle | 16. Gas Control Pressure Indicator |
| 7. Breathing Oxygen Supply Indicator | 17. Jettisoning Handle |
| 8. Rocket Engine Controls | 18. Automatic Flight Control Panel |
| 9. Control Wheel | 19. Misc. Rocket Engine Instruments |
| 10. Rudder Pedals | 20. Panels for Research Instrumentation |



COCKPIT CAPSULE



ESCAPE PROVISIONS IF DEMOUNTABLE
CAPSULE IS INCORPORATED IN AIRCRAFT

12.0 MISCELLANEOUS FLIGHT PROBLEMS

12.1 Accuracy of Flight Variables

The take-off and landing sites for the high altitude flight must be chosen with great care. A landing base of the type of Edwards Air Force Base will be required, with exceptionally long runways and considerable latitude in the choice of direction and position of touchdown. For a typical optimum-altitude flight, the airplane travels a horizontal distance of about 500 nautical miles (Figure 10); since most of this distance is covered in the ballistic trajectory portion of the flight path, there will be little opportunity to control or alter either the range or the heading by any appreciable amount after burnout. During the 75 seconds of powered flight, the pilot will have an opportunity to make necessary corrections to the flight path, although these corrections cannot be very large because of the rapid acceleration. Therefore an accurate positioning and heading of the mother airplane at the time of launching will be important.

The maximum allowable miss distance between the pullout location and the landing site is a function of the respective orientation of the two positions and the maximum gliding distance of the airplane. An indication of the maximum "dispersion area" is shown in Figure 38. The allowable miss distance is much shorter if the airplane overshoots the base because of the necessity for turning around. To give a physical idea of the errors which would cause the maximum dispersions shown, a misalignment of 5° in azimuth at burnout would result in a lateral miss of 40 nautical miles, while a $2\frac{1}{2}$ percent error in fuel consumption or burning time would result in a 60 nautical mile decrement in distance.

It is apparent that great care must be taken with all phases of the flight to insure that the pilot will pull out within this gliding range of the landing site. This is further emphasized when the maneuverability available after recovery to level flight is considered. Assuming that the pilot elects to slow down to subsonic speed at a constant altitude of 40,000 feet, it will then be possible to make no more than six spiral turns before sea level altitude is reached, which gives an indication of the maximum amount of maneuverability available.

12.2 Exploratory Flight Test Procedure

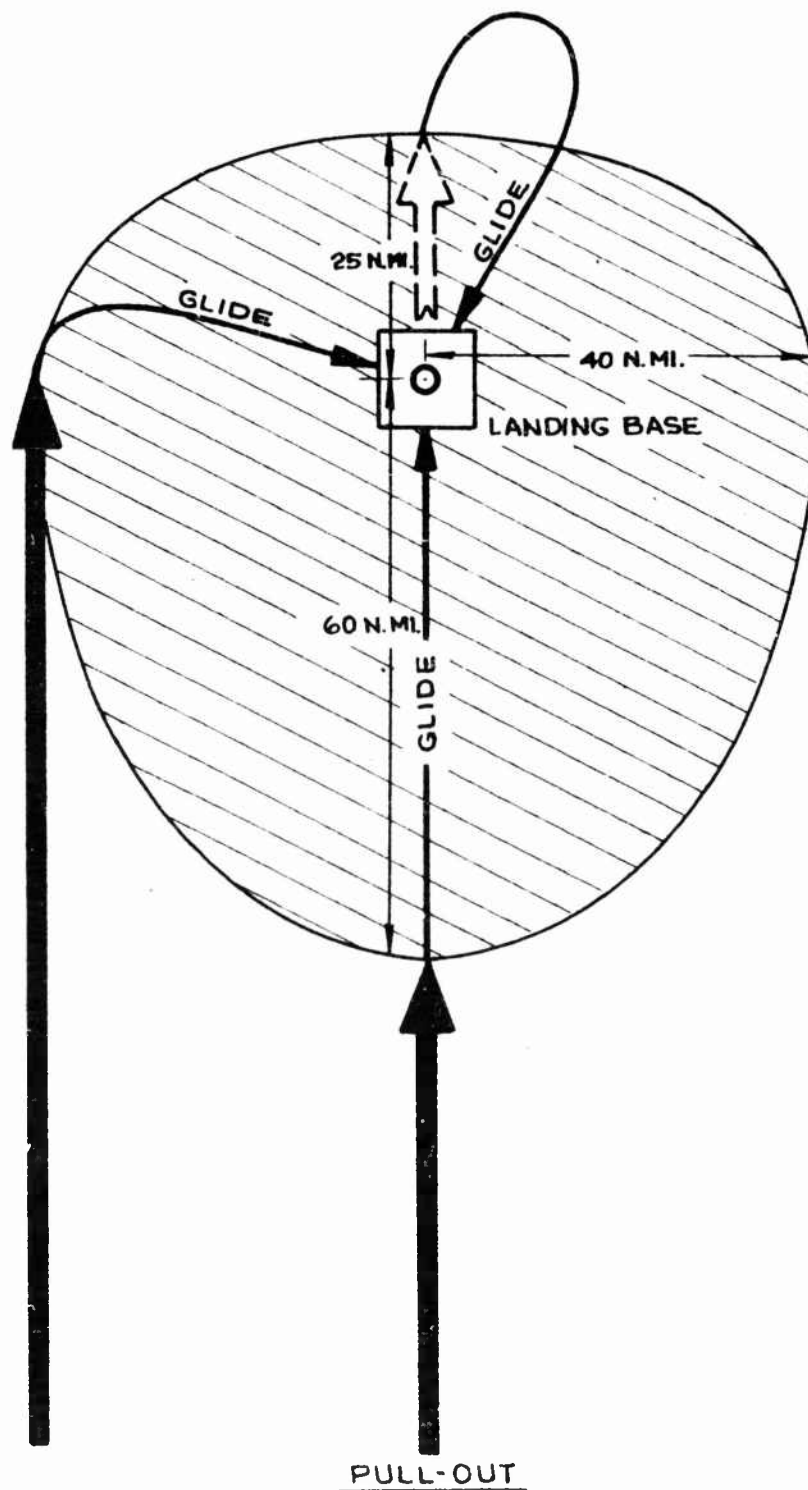
It is apparent that a great number of exploratory flights will be required before any full scale high altitude attempts are made. The first flights would undoubtedly be air-launched glide flights.

These should be followed by flights with low power, either a small rocket motor or perhaps even a turbojet engine. During these preliminary phases of the flight test program, the basic aerodynamic characteristics of the configuration would be evaluated, and individual items of special equipment proven in flight. With installation of the design rocket motor, the flight speeds and altitudes could be improved gradually by increasing the quantity of propellant carried as the test program progressed. In this way all of the problems of temperature, physiology, control, thrust alignment, etc., could be worked out in a systematic manner.

DOUGLAS AIRCRAFT COMPANY, INC. EL SEGUNDO DIVISION EL SEGUNDO, CALIFORNIA

MODEL 671

MAXIMUM DISPERSION AREA
FOR LOCATION AT END OF PULL-OUT



PULL-OUT

13.0 EXTENSIONS

The primary mission of the airplane studied in this report is one of research. The ultimate purpose for which it is designed will extend the frontier of human flight into a regime which is a complete unknown at the present time: the threshold of space.

Before such flights become commonplace, there is a vast quantity of information to be obtained about:

1. Human existence outside of the protection of the earth's atmosphere
2. The effects of high load factors on humans
3. The problem of safe re-entry into the atmosphere
4. The design of airplane structures capable of operation in the presence of extremely high temperatures
5. The problems of automatic control of flight paths

It is expected that an airplane of the type proposed will be capable of providing this information, all of which will be of great value in future military applications.

On the basis of the results of this preliminary study, there appears to be a forbidden region of performance beyond that of the proposed airplane, in which it is impossible to make a safe re-entry into the atmosphere. If this tentative result proves to be valid, then it would appear that the next step following the present design must be a big one, directly to a large, multi-stage, orbital vehicle. This would not necessarily be as large as the satellite type, but it should possess sufficient peripheral speed to permit a shallow re-entry into the atmosphere, with a gradual deceleration as the altitude is decreased. In view of the many unsolved problems which now exist with regard to flight in the upper atmosphere, it seems certain that a more modest vehicle of the type proposed herein will be a necessary forerunner to an orbital or satellite machine.

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